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The Utilization of Supersonic Combustion Ramjet Systems at Low Mach Numbers

Sqn Ldr Edward T. Curran, RAF

Frank D. Stull

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AF Aero Propulsion Laboratory
Research and Technology Division
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

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NOTICES

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FOREWORD

This report was prepared by Sqn Ldr Edward T. Curran, RAF, and Mr. Frank D. Stull, Ramjet Components Branch, Ramjet Engine Division, of the AF Aero Propulsion Laboratory, Research and Technology Division, Wright-Patterson Air Force Base, Ohio. The work described was accomplished under Advanced Technology Program 651E "Supersonic Combustion Ramjet." This work was prepared specifically for presentation at the AIAA Summer Meeting, Los Angeles, California, 17-20 June 1963.

The authors wish to acknowledge the substantial contribution of Mr. Roger R. Craig who generated an appreciable amount of computer data for their use, and the valuable overall assistance of Mr. M. Brian Bergsten, Mr. William E. Supp, Mr. John R. Smith, and Mr. Kenneth Schwartzkopf of the Ramjet Components Branch.

ABSTRACT

Some in-house studies carried out at AF Aero Propulsion Laboratory to investigate the low speed performance of the supersonic combustion ramjet engine is presented. A discussion of the fundamental problems of low speed operation is followed by a review of techniques of converting from supersonic to subsonic combustion. In concluding a new engine concept, the "dual-mode" engine, is discussed, and its potential performance outlined.

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This technical documentary report has been reviewed and is approved.

Ed. A. Hawkens

EDWARD A. HAWKENS,
Colonel, USAF
Director, AF Aero Propulsion Laboratory

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LIST OF SYMBOLS

A	Area
C_v	Velocity Coefficient of Nozzle
h	Enthalpy
I_{sp}	Fuel Specific Impulse
K_D	Intake Process Efficiency
M	Mach Number
n	Polytropic Exponent
P	Static Pressure (psia)
q	Dynamic Pressure
T	Static Temperature (°R)
V	Velocity
V_F	Velocity of Injected Fuel
γ	Ratio of Specific Heats
δ	Deflection Angle
η_c	Combustion Efficiency
η_{KE}	Kinetic Energy Efficiency of Inlet
ρ	Density
Φ	Equivalence Ratio

Subscripts $_0$, $_2$, $_3$, and $_4$, refer to the reference stations shown in Figure 2

Subscript $_*$ refers to the sonic condition

INTRODUCTION

The supersonic combustion ramjet engine (Scramjet) has been shown, in Reference 1, to possess the potential of high performance in the speed range Mach 8-25. However, operation of such engines at speeds substantially below Mach 8 has received little attention previously. Presumably this is because of two factors: (1) it is generally conceded on the basis of current component efficiencies, that subsonic combustion engines yield better performance at the lower speeds; and (2) the extension of Scramjet operation to lower speeds appears inherently difficult. However, acceleration of a Scramjet powered vehicle to the region of Mach 8 is a formidable problem. Acceleration to this speed by a turbo-accelerator engine is questionable unless a complex multi-mode operation is undertaken. It follows, therefore, that any significant reduction in the take-over speed of the Scramjet could provide a major simplification in the turbo-accelerator system with considerable benefit to the overall vehicle performance. Even if supersonic combustion cannot itself be sustained at low Mach numbers, every effort should be made to utilize the main Scramjet duct for efficient propulsion at these lower speeds. For example the Scramjet duct might be used either as a subsonic combustion ramjet or as a mixing chamber for a thrust augmentation device of the ram-rocket type. Obviously such utilization must not seriously degrade the Scramjet performance at the higher Mach numbers.

We aim in this report to stimulate thought about methods of utilizing the Scramjet duct at low speeds to obtain efficient propulsion. As a possible end product, one can conceive a fixed geometry propulsion system which would operate over a wide Mach number range, say from 4 through 25. As an intermediate goal a fixed geometry system operating through a more restricted Mach number range, say from 2 to 12, would be an attractive engine for hypersonic flight applications.

In this report we are concerned essentially with engine/vehicle operation in the Mach number range of 3 to 10. A typical vehicle is illustrated in Figure 1. The corresponding stations used in cycle calculations are shown in Figure 2. The trajectory considered is generally limited to a dynamic pressure of 1250 psf.

DESIGN FEATURES OF A HIGH SPEED ENGINE

Before considering the problems of operating a Scramjet at low speeds it is necessary to consider the design features of an engine designed for high Mach number operation. One of the major features is the shape of the combustor chamber and the associated mode of heat addition. Most project work is at present based on either constant area or constant pressure heat addition processes. However, irrespective of the choice of constant area or constant pressure processes, there is very little difference in the performance levels achieved at the higher hypersonic speeds.

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A performance chart for a Scramjet with a constant pressure combustor and operating at a flight speed of 26,000 ft/sec is shown in Figure 3. In Figure 4 the corresponding area ratios for a possible engine design are shown; the intake capture area ratio is of the order of 20-50:1, and the combustor area ratio is approximately 2:1. For the type of vehicle under consideration the maximum nozzle area will be approximately equal to the inlet capture area; thus the nozzle area ratio will be about 10-25:1.

LOW SPEED PERFORMANCE OF SCRAMJET

Limitations on Heat Addition

As the flight speed of the engine is reduced, efficient operation requires that more diffusion be performed by the intake. Thus the cumulative effects of reducing flight speed and increasing diffusion result in a severe reduction in the Mach number, at entry to the combustor. This decrease has serious consequences for combustor operation. In the case of a constant area combustor operating with a given supersonic entry Mach number, the amount of heat which can be added to the flow is limited by the attainment of sonic speed at the combustor exit. Any further attempt to add heat to this thermally-choked flow will result in shock wave formation upstream of the combustor and breakdown of the supersonic flow at the entry to the combustor. Thus as the combustor entry Mach number is reduced the amount of heat which can be added to the flow is reduced. This limitation is shown in Figure 5 for the combustion of hydrogen and air. The allowable equivalence ratio is plotted as a function of flight speed and intake diffusion. Thus a rapid fall in equivalence ratio occurs as both flight speed and velocity ratio decrease. This reduction in equivalence ratio is reflected in a considerable loss in the specific thrust of the engine.

Turning now to the case of the constant pressure combustor, which also operates at constant flow velocity, the thermal choking phenomenon does not occur. For a given supersonic Mach number at entry to the combustor the continuing addition of heat will eventually reduce the flow Mach number to a subsonic value, the combustor area continuously increasing with heat addition. However, it is considered that the attainment of sonic flow at the combustor exit is still a practical limitation on the amount of heat addition because, if the flow is allowed to go subsonic, then some form of convergent-divergent exit nozzle will be required to expand the flow. The equivalence ratio required to produce a sonic condition at the exit of a constant pressure combustor is shown in Figure 6. This limitation on the equivalence ratio is not as severe as for the constant-area case shown in Figure 5. It should be noted that all the above limits are based on one dimensional considerations and include real gas efforts. Fuel injection effects have been neglected.

Various supersonic heat addition processes were considered based on perfect gas relationships. The area ratios required for these are shown in Figure 7. Except for the constant static temperature and constant Mach number processes, both the Mach number at the combustor entrance and combustor exit were fixed at 6 and 1.1, respectively. The lines, representing these processes, end at a total temperature ratio, T_{o3}/T_{o2}, corresponding to the amount of heat required to reach Mach 1.1 at the combustor exit. Results are indicated by the circles where the combustor inlet Mach number was changed from 6.0 to 2.5, thereby allowing for diffusion in the inlet at a flight Mach number of 6.0. Note that the polytropic processes with a positive exponent represent convergent ducts and can take very

little heat addition before reaching near sonic conditions. For the constant Mach number process and the constant static temperature process, both the prescribed boundary conditions cannot be fulfilled; the combustor entrance and exit Mach are by definition equal in the constant Mach number process, and in the case of the constant static temperature the exit Mach number increases rather than decreases with heat addition. For a given flight Mach number the approximate fuel equivalence ratio, Φ, may be related to the total temperature ratio as indicated in the figures. It is interesting that for reasonable combustor area ratios (less than 5) at an equivalence ratio of one, the constant pressure process is very attractive. Only the performance of the constant pressure combustion Scramjet will be presented herein.

Cycle Performance at Low Speeds

For the purposes of estimating performance at low speeds the following engine conditions were assumed:

Intake kinetic energy efficiency	η_{KE}	= 98, 99, & 100 percent
Combustion efficiency	η_c	= 100 percent
Nozzle velocity coefficient	C_v	= 0.99
Nozzle area ratio	A_4/A_0	= 1.00

Fuel injected normal to flow and at 37°R

Equilibrium conditions

Specific impulse, at Mach 6.0 for 3 different Φ's, plotted against V_2/V_0 is presented in Figure 8. Actually this form of presentation, while valuable for stoichiometric conditions, is somewhat misleading when low values of Φ are encountered. Although I_{sp} increases with decreasing Φ's, the thrust coefficient falls off rapidly thereby giving poor overall performance. As a result a new form of presentation is adopted herein.

Some representative results (for an η_{KE} of 0.98) are shown in Figures 9, 10, and 11, corresponding to Mach numbers of 4.0, 5.0, 6.0. In each figure the performance of the engine is represented by the parameter, $\Phi\ I_{sp}$, which is shown as a function of Φ and V_2/V_0. By using the parameter, $\Phi\ I_{sp}$, it is a simple matter to assess both the I_{sp} and specific thrust of the engine, as the latter is proportional to the product Φ and I_{sp}. On each chart the equivalence ratio limitation, due to the sonic exit condition, is indicated. This limitation is not significant at Mach 6.0 but becomes very restrictive at Mach 4.0.

As presented, these performance curves relate to variable geometry engines. It is pertinent to illustrate next the effect of fixed geometry inlet operation on the attainable cycle performance.

Off-design Operation

From examination of Figure 4 it is apparent that at very high speeds the diffusion performed by the inlet is very small ($0.95 < \frac{V_2}{V_0} < 1$). The type of inlet geometry resulting from such consideration will usually consist of a low angle ramp followed by additional isentropic compression. However, for purposes of illustrating intake characteristics at the lower speeds, a simple two shock intake will initially be considered in this report. The performance of such an inlet is shown in Figure 12, where η_{KE} and V_2/V_0 are shown as functions of flight Mach number and wedge angle.

It is generally accepted that for a given Mach number a given inlet will operate at a substantially constant process efficiency (K_D) as V_2/V_0 varies, that is, η_{KE} is related to V_2/V_0 by the relation $\eta_{KE} = K_D (1-K_D) (\frac{V_2}{V_0})^2$. However, for a given inlet configuration it appears that the effect of reducing flight speed is to increase the diffusion performed by the inlet and simultaneously to increase kinetic energy efficiency. In fact for this inlet the trend is toward operation at constant η_{KE} at the lower Mach numbers. The precise variation of the inlet parameters; such as η_{KE}, V_2/V_0, pre-entry drag, and spillage, cannot be presented in parametric terms at this time due to the lack of inlet design information for the speed range considered. Additionally, it is difficult to generalize such information because of the close integration of engine and vehicle design. For example, the performance of a low angle wedge inlet is considerably affected by the incidence schedule flown by the vehicle, which may be determined largely from engine considerations. An additional complication concerns the changes in the internal shock reflections within the inlet which arise as the flight speed is reduced. Exact cancellation of the reflected shocks may only be anticipated at the design condition. At off-design conditions the reflected shocks may not comply with the necessary flow boundary conditions, and near-normal shocks may be generated with complete flow breakdown.

Returning to the simple two shock inlet considered in Figure 12, it is evident from the performance shown in Figures 9 to 11 that the diffusion provided by this simple intake is quite inadequate for acceptable low speed performance. Thus for good performance at low speeds it is necessary to design an inlet which will accomplish significantly more diffusion than the simple two shock inlet. In the present state of the art only one known example of a fixed geometry inlet yielding the diffusion characteristics required at low speeds (Mach 6 in this case), yet maintaining good performance at high speeds (Mach 25), has appeared. This inlet is the one described in Reference 2.

In view of the limited amount of inlet data it is essential to postulate some type of inlet performance. We assumed that the intake performance is described by the following:

1. The intake operates at a constant velocity decrement, that is, $V_0 - V_2$ = constant. This assumption establishes the velocity ratio schedule as a function of V_0.

2. The inlet operates at a constant value of η_{KE}, this value being dependent on the velocity decrement* associated with the inlet.

*Some interesting computations using the velocity decrement parametric are given in Reference 3.

3. The spillage characteristic of the inlet is consistent with the above two postulates, and thus inlet choking is avoided.

By way of illustration the velocity schedules corresponding to various assumed values of the velocity decrement are shown in Figure 13 together with some available inlet data. The variations of inlet enthalpy ratio and K_D with Mach number are shown in Figures 14 and 15.

Discussion

The effect of the intake diffusion characteristics on Scramjet performance is shown in Figure 16. For a given inlet velocity decrement, stoichiometric operation is maintained as flight speed is reduced until the sonic exit condition is reached. With further reduction in flight speed the equivalence ratio is reduced with a consequent reduction in specific thrust and an initial rise in specific impulse.

With higher amounts of diffusion, for example, V_o - V_2 = 2000, the overall performance of the engine is improved; but, the sonic exit limit is encountered at higher flight speeds than it is with lower amounts of diffusion. In general, however, the cycle performance deteriorates quite rapidly below about Mach 5.0. If more restrictive assumptions, concerning pre-entry drag or fixed nozzle performance, were incorporated into this analysis then the performance would deteriorate even more at the lower speeds.

The degree of inlet diffusion also exerts an influence on the area ratios of the constant-pressure combustor. This effect is shown in Figure 17 for an equivalence ratio of one. With low amounts of diffusion the temperature ratio across the combustor is high and the area ratio is correspondingly large. With increasing flight speed the area ratio decreases; however, this should not be taken to imply that a variable area combustor is required. Most probably a combustor could be designed for the highest area ratio required at the lower speeds, and an inefficient but acceptable expansion process be allowed to commence in the chamber at the higher flight speeds. An attractive alternate to avoid variable area would be to schedule the fuel flow as a function of flight speed. Typical fuel schedules are shown in Figure 18 for an intake ΔV of 1500.

Up to this stage one major problem of low speed operation has been purposely avoided, namely the effect of low speed operation on the chemical kinetics of the combustion process. The effects of static pressure and temperature on the time required for the overall reaction of hydrogen and air are shown in Figure 19. The reaction time has been calculated from the empirical equations of Reference 4. For short reaction times temperatures in excess of 2000°R and pressures greater than about 5 psia are required. In low speed operation comparatively low temperatures are obtained, and despite the lower flow velocities the overall reaction times are generally excessive. This situation is illustrated in Figure 20 where the static temperature at entry to the combustor is shown for various levels of diffusion. It is probable that supersonic combustion will not be sustained at local static temperatures much below 1800°R. It can be seen that the intakes with the highest diffusion will yield acceptable conditions down to about Mach 6.0. However, it is anticipated that chemical kinetic conditions will be marginal during low speed operation and additional flame stabilizing devices will be required. Such devices should not adversely affect the high speed operation of the engine. Some suggested piloting devices are shown in Figure 21.

These schemes are largely self explanatory although the induced shock pilot deserves some comment. In this scheme the shock in the annular duct is swallowed at the higher Mach numbers; at lower speeds the shock "pops" creating a local region of extensive subsonic flow at elevated pressure and temperature. The wall recessed pilot may assume more predominance than in the past since hydrogen will be available to actively cool this configuration. These various schemes may well overcome the chemical kinetic problems associated with low speed operation. Alternatively the chemical kinetic problems associated with hydrogen and air may be avoided by utilizing pyrophoric fuels. No attempt has been made to assess the potential advantages of using such highly reactive fuels. The major problem areas discussed are summarized below.

1. Wide Mach number operation of fixed geometry components

2. Limitations on heat addition in constant area and constant pressure ducts

3. The importance of obtaining adequate diffusion at the lower speeds

4. Chemical kinetic problems of combustion

It appears to be difficult to achieve adequate performance at very low speeds such as Mach 2 to 4. Furthermore, since the subsonic burning ramjet provides better overall performance at these lower speeds it is worth investigating the possibility of converting the Scramjet to the subsonic burning mode. The rest of this report will be concerned with this problem.

SUBSONIC-SUPERSONIC CONVERSION

In considering the type of heat addition mode for the lower speeds, a preliminary survey was conducted of engines employing both oblique and strong detonation waves (Figure 22). Because of the uncertain and limited performance of such engines it was decided to restrict this work to considerations of only the conventional subsonic burning ramjet engine. A typical performance chart for such an engine is shown in Figure 23. For a high thrust engine such as required for acceleration missions, acceptable performance is obtained with equivalence ratios approaching 1.0 and with small degrees of nozzling, that is, $A^*/A_c = 0.9$ even though a small penalty is encountered in I_{sp}.

It appears that one of the first requirements in converting to the subsonic mode is to provide a small degree of contraction following the combustor. Various methods of providing this contraction may be considered as shown in Figure 24.

It may be possible to provide a small permanent degree of contraction without seriously impairing the supersonic-combustion performance of the engine. However, the requirement for a convergent nozzle section may not be stringent from the performance viewpoint since acceptable engine performance can be obtained with a straightpipe combustor. A rather more important requirement is to reduce the divergent area ratio at lower speeds to prevent overexpansion losses; two simple 2-position schemes are sketched in Figure 24.

The most central problem is that of achieving a controlled conversion between the subsonic and supersonic modes with a continuous thrust output. One method of achieving such conversion is by means of a mixed-flow engine system. In its mechanical form such an engine is illustrated in Figure 25. During high speed operation the pilot zone operates with supersonic flow; at a lower speed a normal shock is disgorged ahead of the pilot creating an extensive subsonic flow region. Such effects may also be obtained by aerodynamically creating mixed flows. Naturally the whole transition process could be controlled by using a variable geometry system if a variable geometry combustor proved practicable. However, it must be noted that regardless of how the transition from supersonic to subsonic conversion is accomplished, provisions must be made for matching the inlet to the combustor. This problem is illustrated in Figure 26 where the required capture area ratios for typical supersonic and subsonic-combustion conditions are shown. If transition from one mode of combustion to another is made at any given Mach number then some method of adjusting the inlet capture area ratio is required. The most efficient means of changing the capture area ratio is by utilizing variable geometry. Another method proposed is the "dual-mode combustor" and is illustrated in Figure 27; in this case the inlet is designed for the supersonic mode but is allowed to operate in the subsonic mode as a supercritical conventional intake. The special feature of this engine system is that two combustors are provided, one for supersonic combustion and the other for subsonic operation. The supersonic combustion precedes the subsonic one and acts as the subsonic diffuser of the intake during subsonic-combustion. The transition from one mode to another is illustrated schematically in Figure 28.

It will be seen that in the subsonic mode fuel is injected into only the subsonic combustor, the flame being stabilized on recessed flameholders. The intake operates supercritically. To make the transition to supersonic combustion, fuel is injected upstream of the swallowed shock, and at the same time, the fuel flow to the subsonic combustor reduced. Thus the shock moves downstream and is ejected from the engine. Finally all the fuel is injected into the supersonic combustor and full transition is achieved. The process described above yields a steady transition from subsonic to supersonic combustion in a fixed geometry system. Although separate combustors and injectors have been shown in this illustration it is probably that an integrated engine system using one set of injectors could be developed. This "dual-mode" engine scheme may be one practical solution to the problem of efficient lowspeed operation.

It is interesting that Perchonok (Reference 5) has demonstrated a controllable and stable transition from supersonic to subsonic combustion, and vice-versa, in a constant area duct.

To assess the cycle performance of the dual-mode engine in the subsonic burning mode four typical configurations were investigated corresponding to intake velocity decrements of 1000 and 1500 and combustor area ratios of 3 and 4. The results are shown in Figure 29. In general the engine performance maximizes at about Mach 4.0. Below Mach 4.0 the over-expansion losses in the nozzle cause a sharp loss in performance. Above Mach 4.0 the increasingly supercritical operation of the inlet also brings about a deterioration in performance as the Mach number increases.

It is emphasized that the performance shown in Figure 29 is the cycle performance. No allowance has been made for the pre-entry drag associated with the inlet or for the effects of the substantial spillage which occurs at the lower speeds. However, for the low angle ramps associated with this class of inlet the pre-entry drag is quite small. Calculations including the effect of spillage have not been made due to the lack of representative data in this area.

Bearing in mind then the somewhat optimistic nature of the performance results obtained for the subsonic operating mode, it is interesting to consider Figure 30, where the performance corresponding to both modes of operation is shown for two values of inlet diffusion. For the inlet with the larger amount of diffusion the overall level of performance is higher for speeds in excess of Mach 3.7. If lower inlet diffusion is accepted then the performance at the higher speeds is impaired, but the low speed performance is improved. In either case it appears that this dual mode engine yields good cycle performance at Mach 3 and, with conversion to supersonic combustion in the region Mach 6 to 7, will continue to yield acceptable performance out to very high hypersonic Mach numbers.

It will be apparent that if such an engine system is specifically designed for wide Mach number operation many design parameters, such as the intake and combustor area ratios, can be chosen to optimize overall engine performance. Inlet spillage characteristics will be of particular importance and must be carefully considered. Such an engine system would appear to be very attractive for hypersonic flight applications such as pure boost/cruise missions.

A particularly attractive by-product of this concept would be to use such an engine in a hypersonic test vehicle for exploration of flight at speeds in excess of Mach 8. The engine itself and its conversion process could be investigated in available ground test facilities so that a reasonable level of confidence could be gained before flight test. Once initial flight testing at the lower speeds, using subsonic combustion, was completed, exploratory investigations of the region beyond Mach 8 could be undertaken with the engine operating in the supersonic combustion mode. In this manner a sound technological base for hypersonic flight could be established. Such a program would not have to await the development of hypersonic turboramjets or turbo-accelerators since the low take-over Mach number associated with the dual mode engine should be within the capability of current technology.

CONCLUSIONS

This investigation of the low speed performance of supersonic combustion ramjet engines has shown that such performance is largely limited by the following factors:

1. The diffusion schedule of the intake

2. The limited amount of heat which can be added to a one-dimensional supersonic stream without encountering choking or a sonic exit condition

3. The chemical kinetic limitations imposed by the low temperature environment

Although the limitations imposed by the above factors can be overcome to some degree by various design features, it is anticipated that conversion to subsonic burning will be required for operation at low supersonic Mach numbers. Two possible schemes have been briefly mentioned, namely the mixed-flow engine and the dual-mode combustion engine. The latter engine has been shown to possess good cycle performance over a wide speed range; but much further work is necessary to assess the performance of typical fixed geometry design.

It is finally concluded that the further study of such convertible subsonic-supersonic schemes is highly desirable, since there are many interesting applications to hypersonic vehicles.

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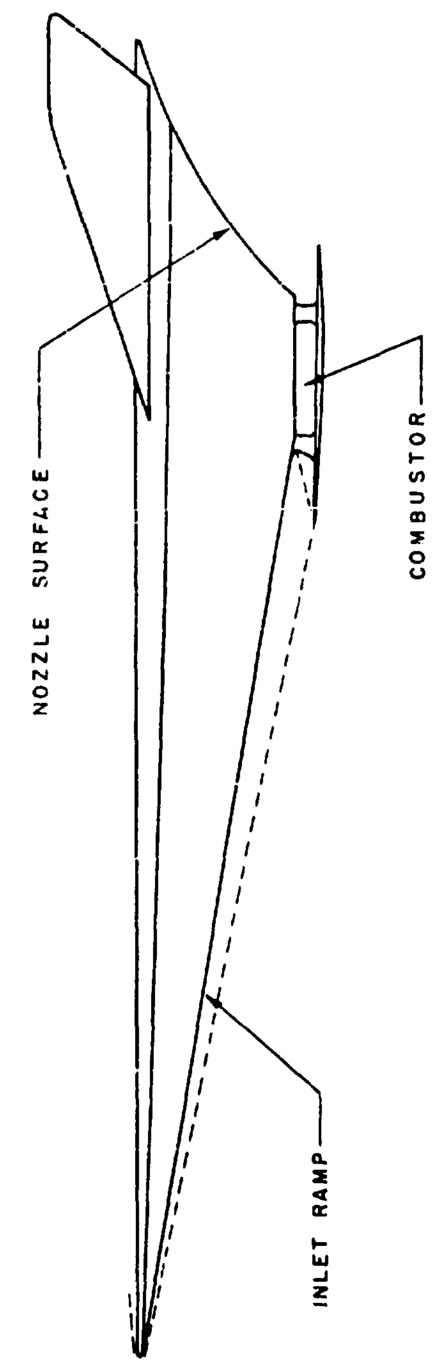

Figure 1. Schematic Representation of An Integrated Aerospace Vehicle

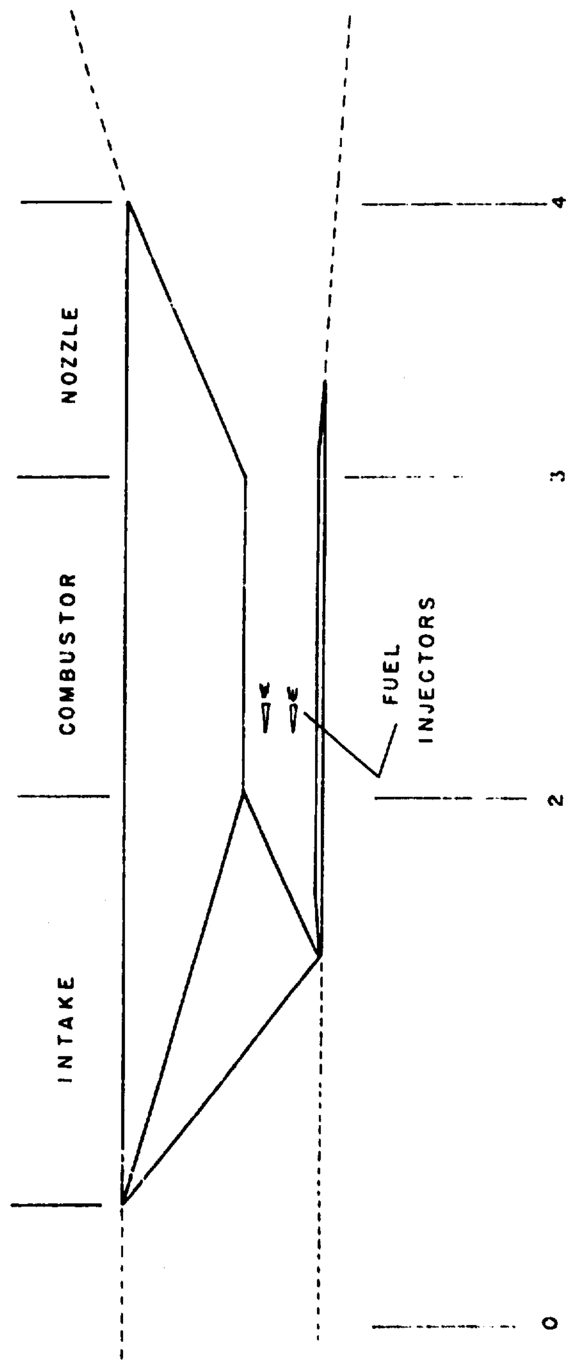

Figure 2. Schematic Arrangement of Supersonic Combustion Ramjet Engine

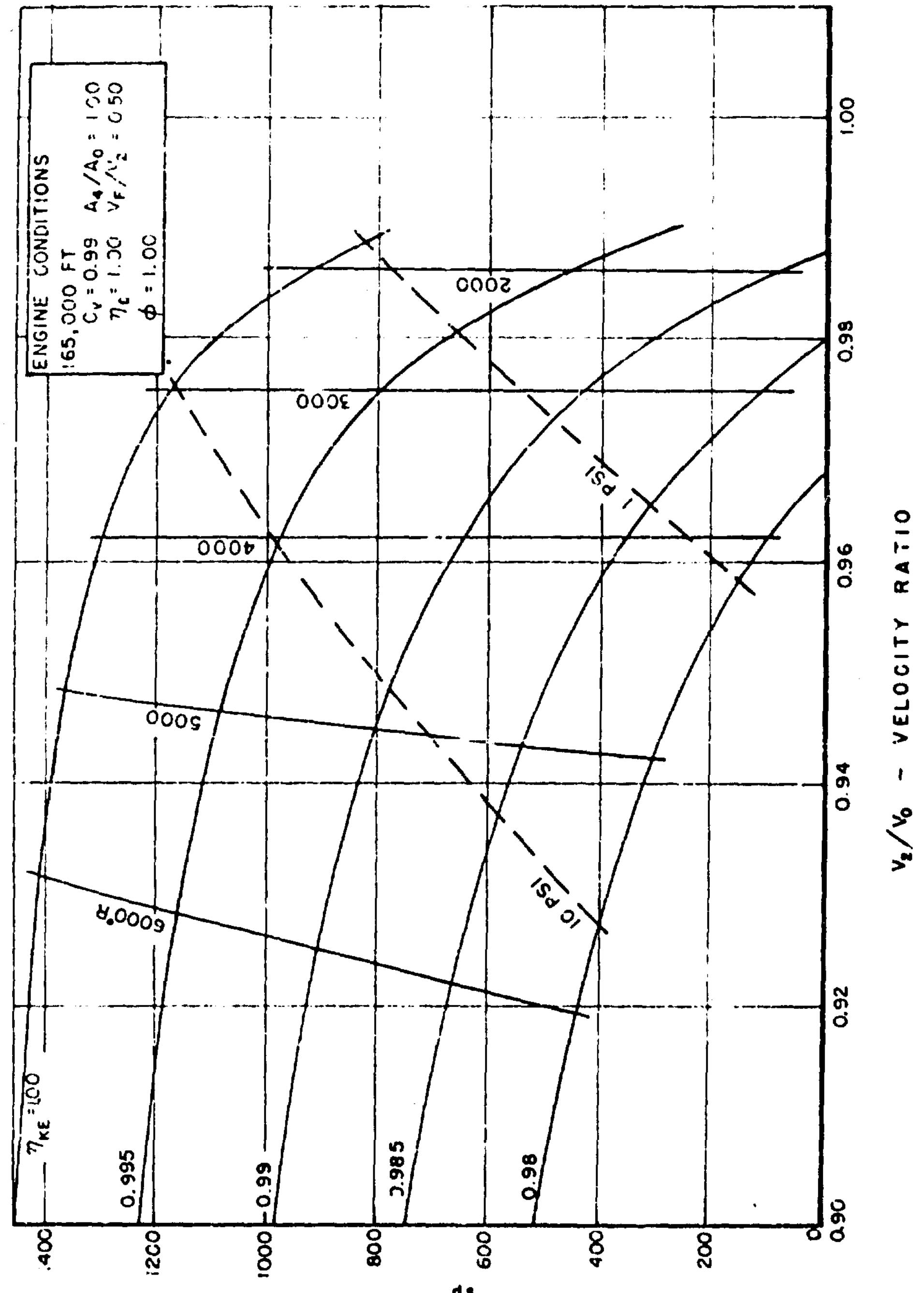

Figure 3. Performance of Constant-Pressure Combustion Scramjet at 26,000 Ft/Sec

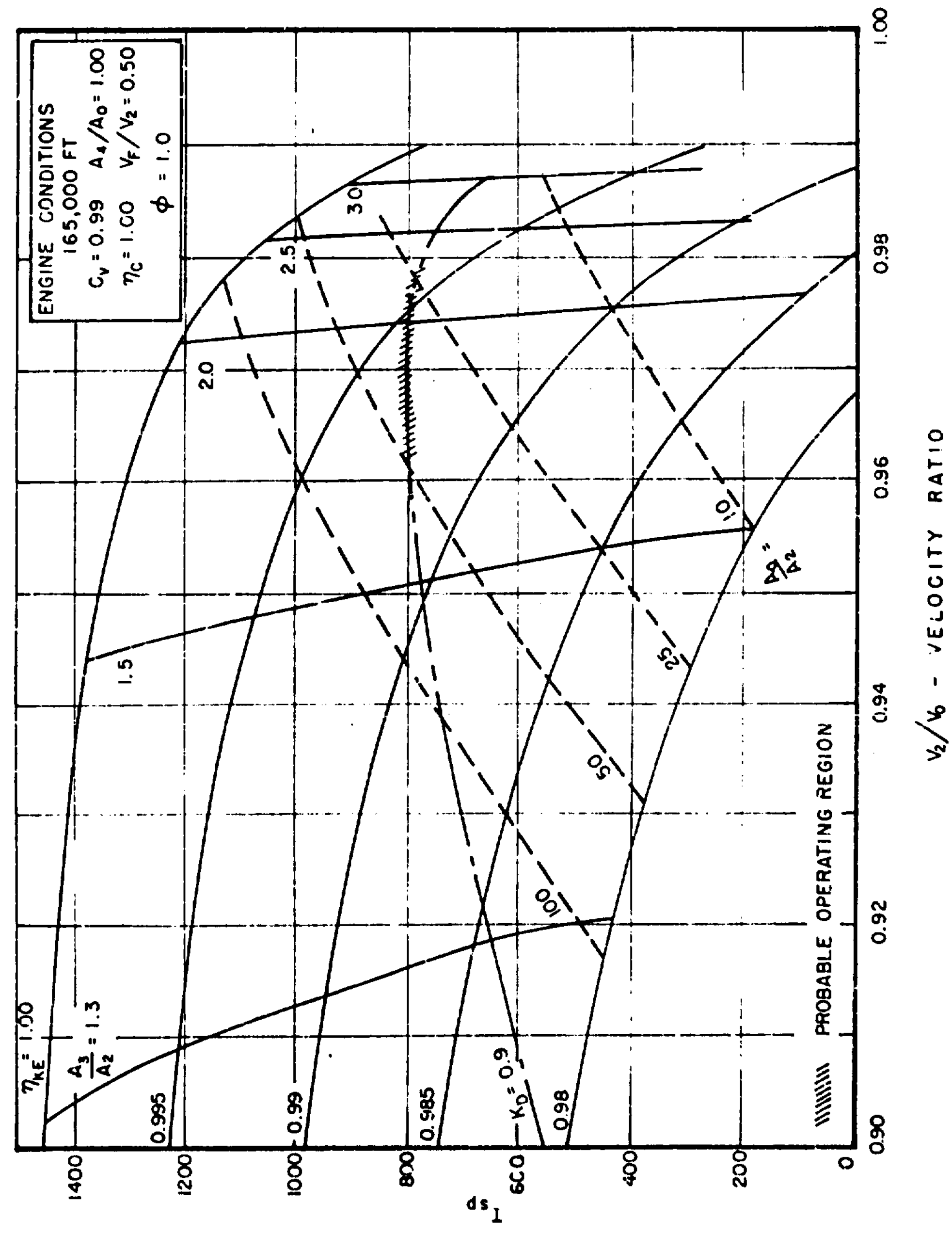

Figure 4. Performance of Constant-Pressure Combustion Scramjet at 26,000 Ft/Sec

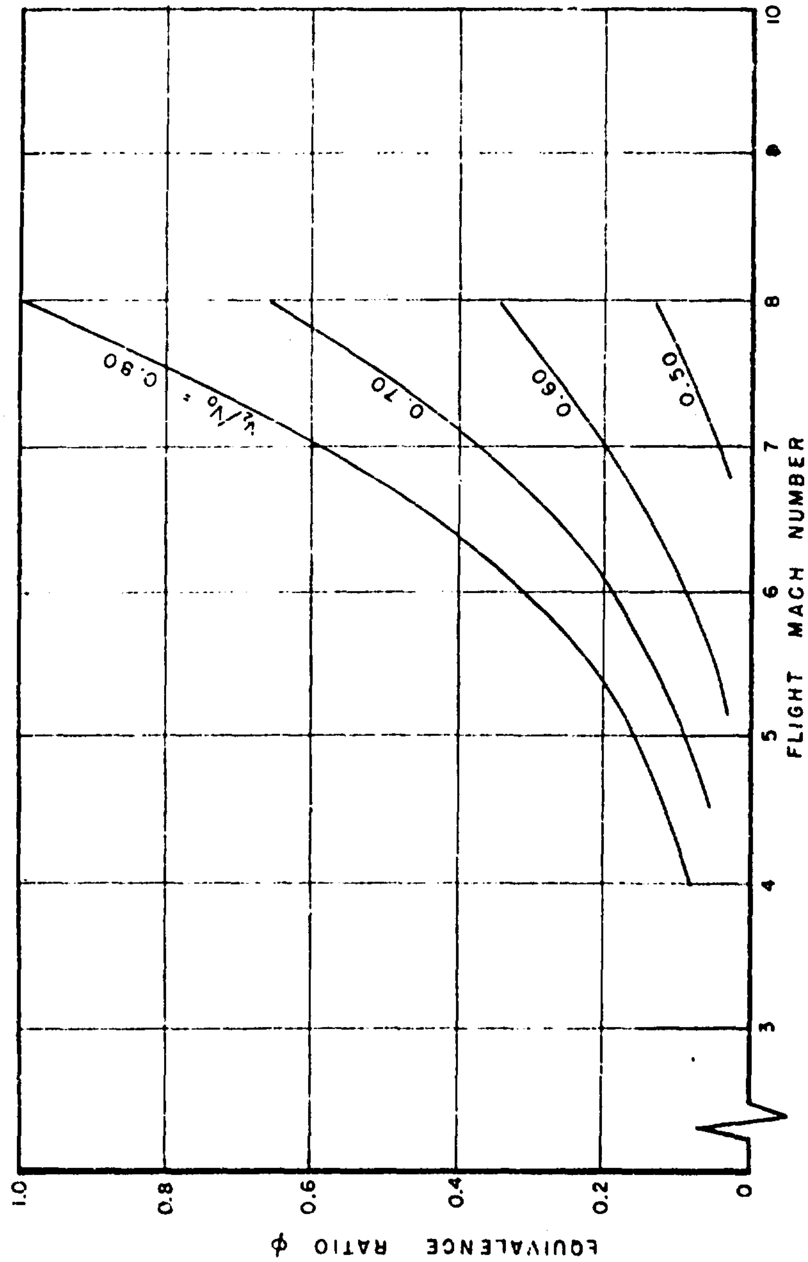

Figure 5. Choking Limits for Constant-Area Combustion

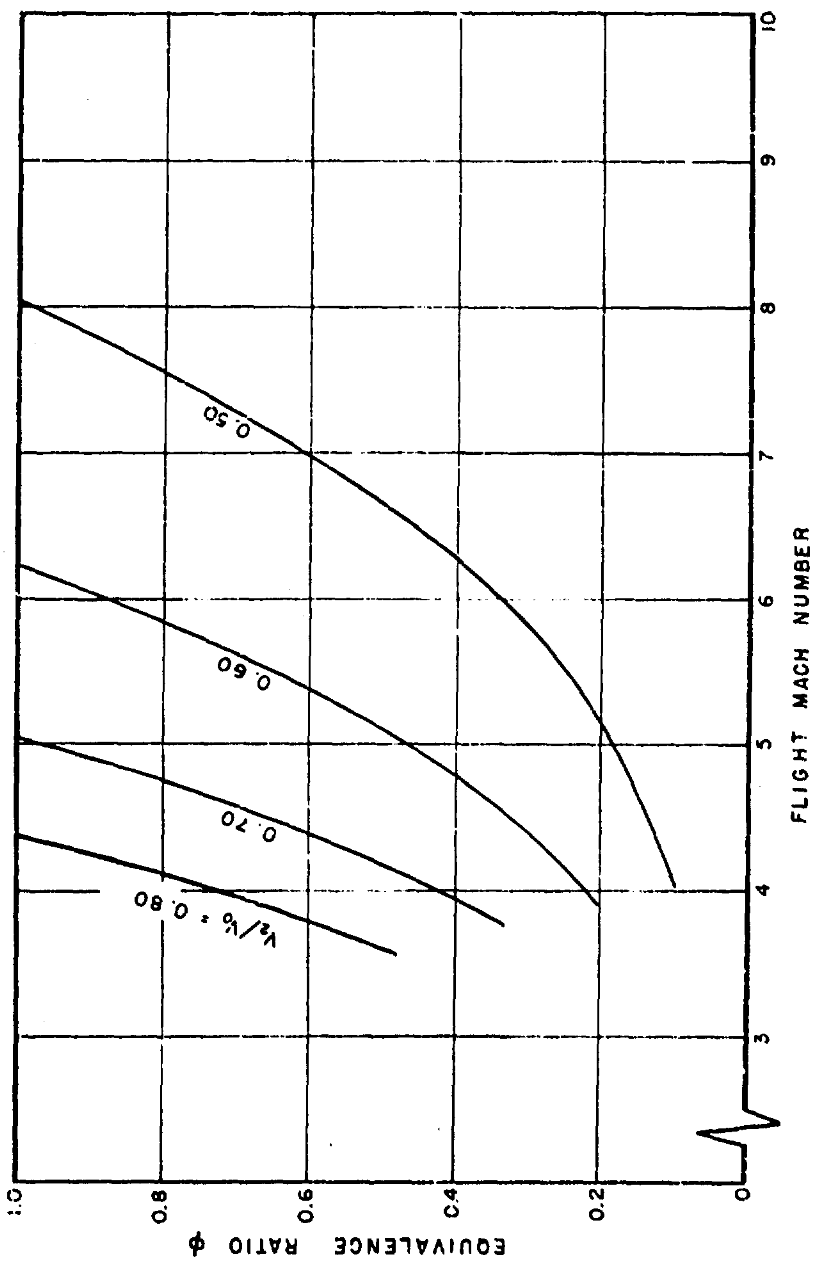

Figure 6. Sonic Exit Limit for Constant-Pressure Combustion

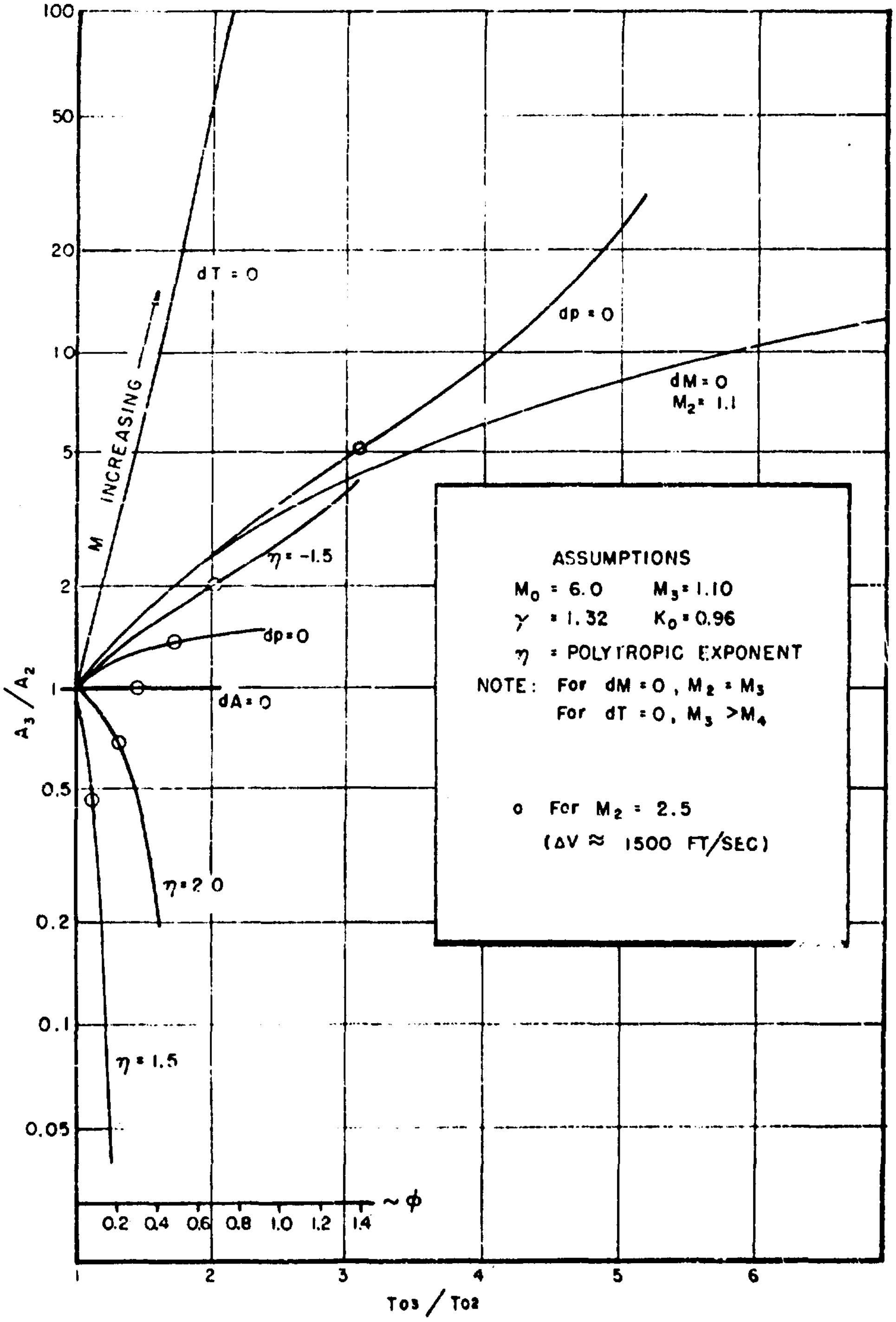

Figure 7. Effect of Various Heat Addition Processes at Low Mach Number on A_3/A_2

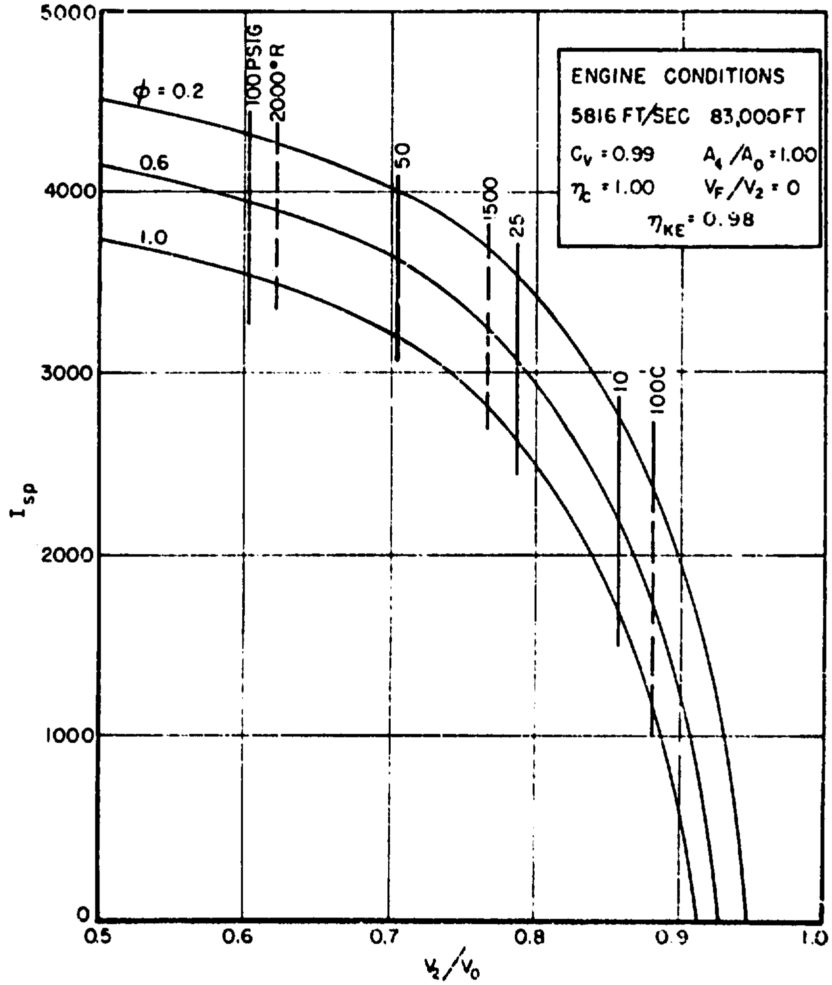

Figure 8. Engine Performance at Mach 6.0

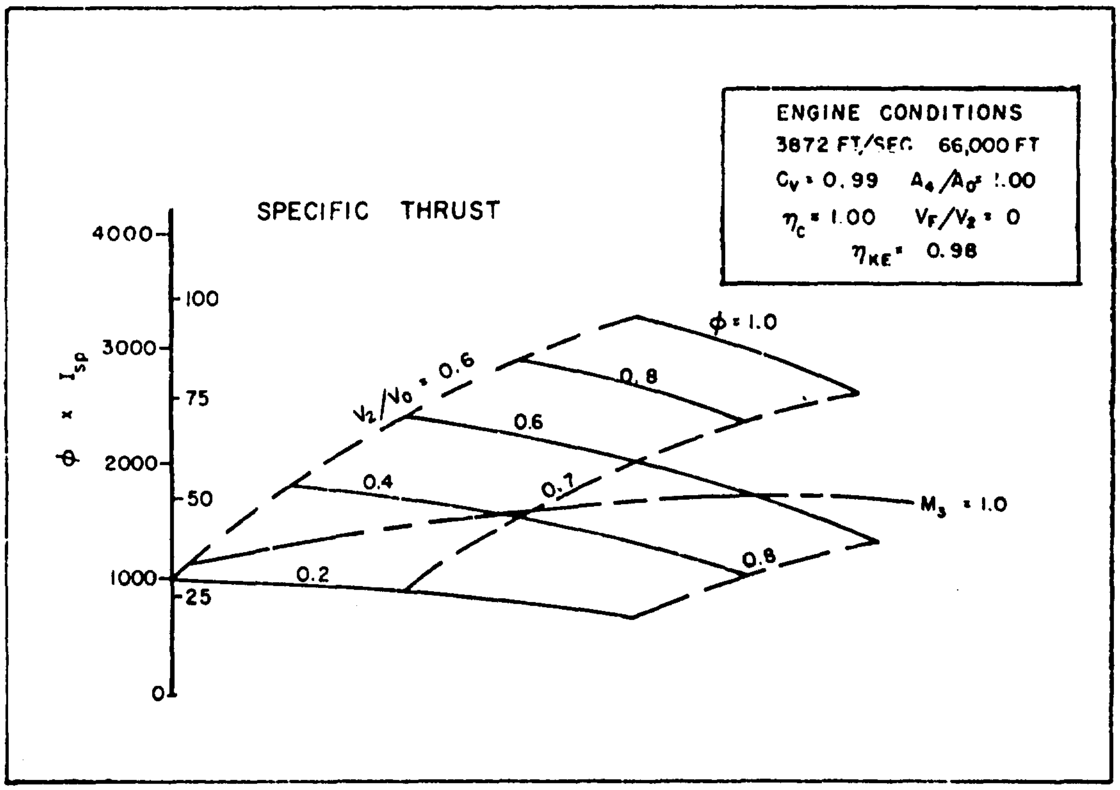

Figure 9. Scramjet Performance at Mach 4.0

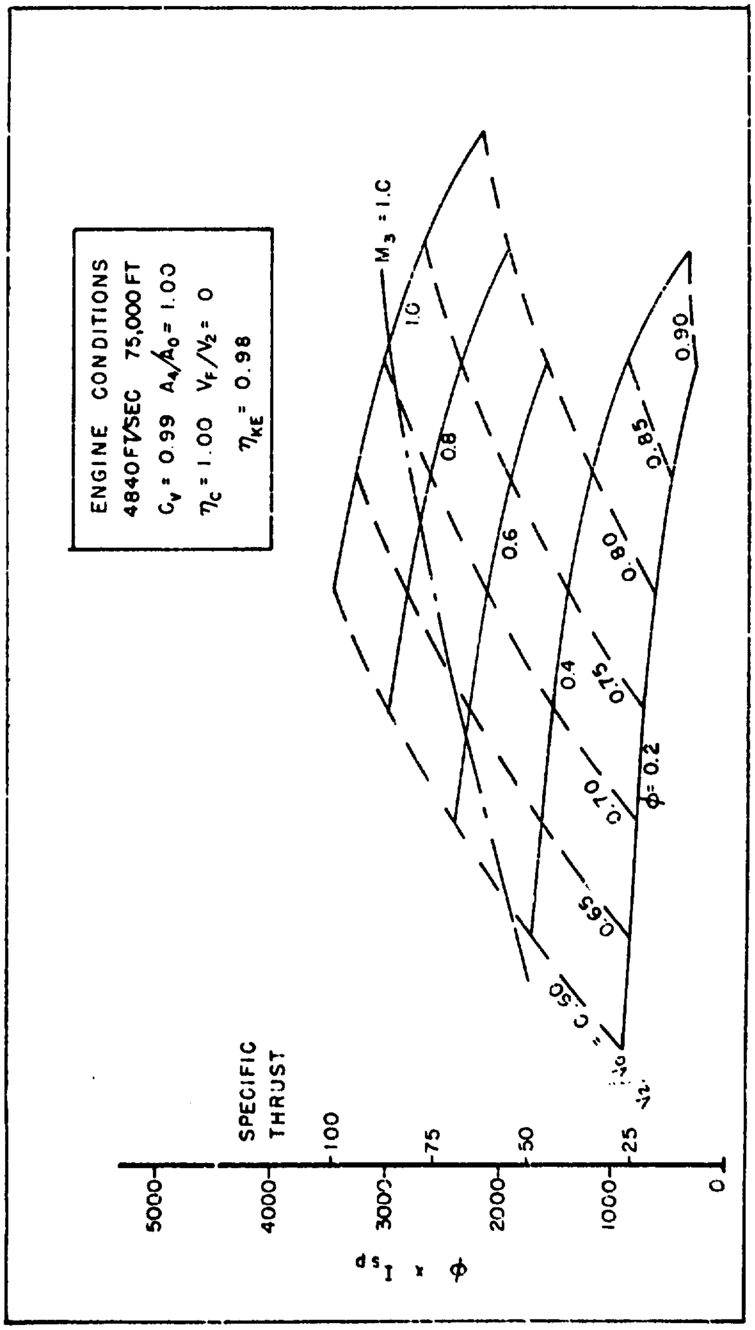

Figure 10. Scramjet Performance at Mach 5.0

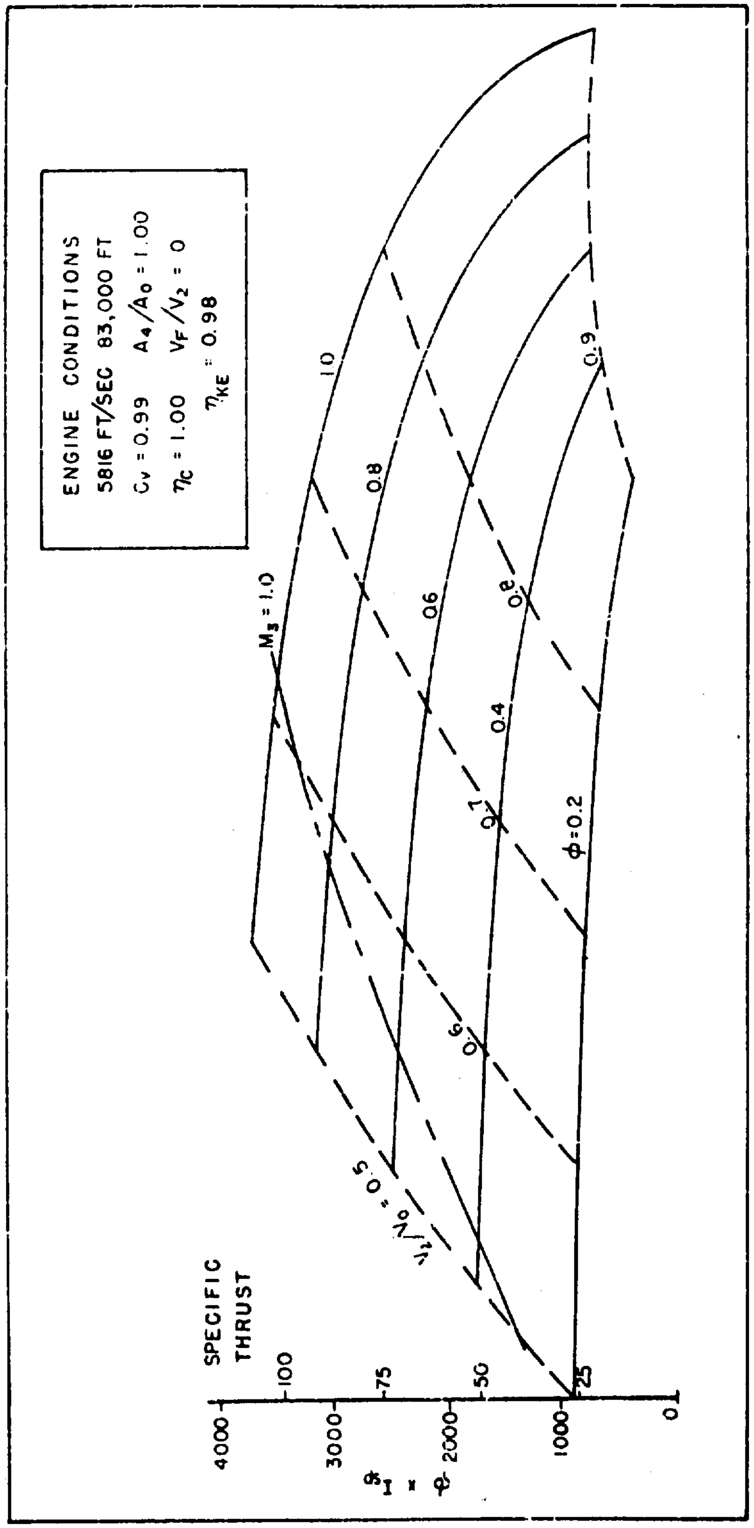

Figure 11. Scramjet Performance at Mach 6.0

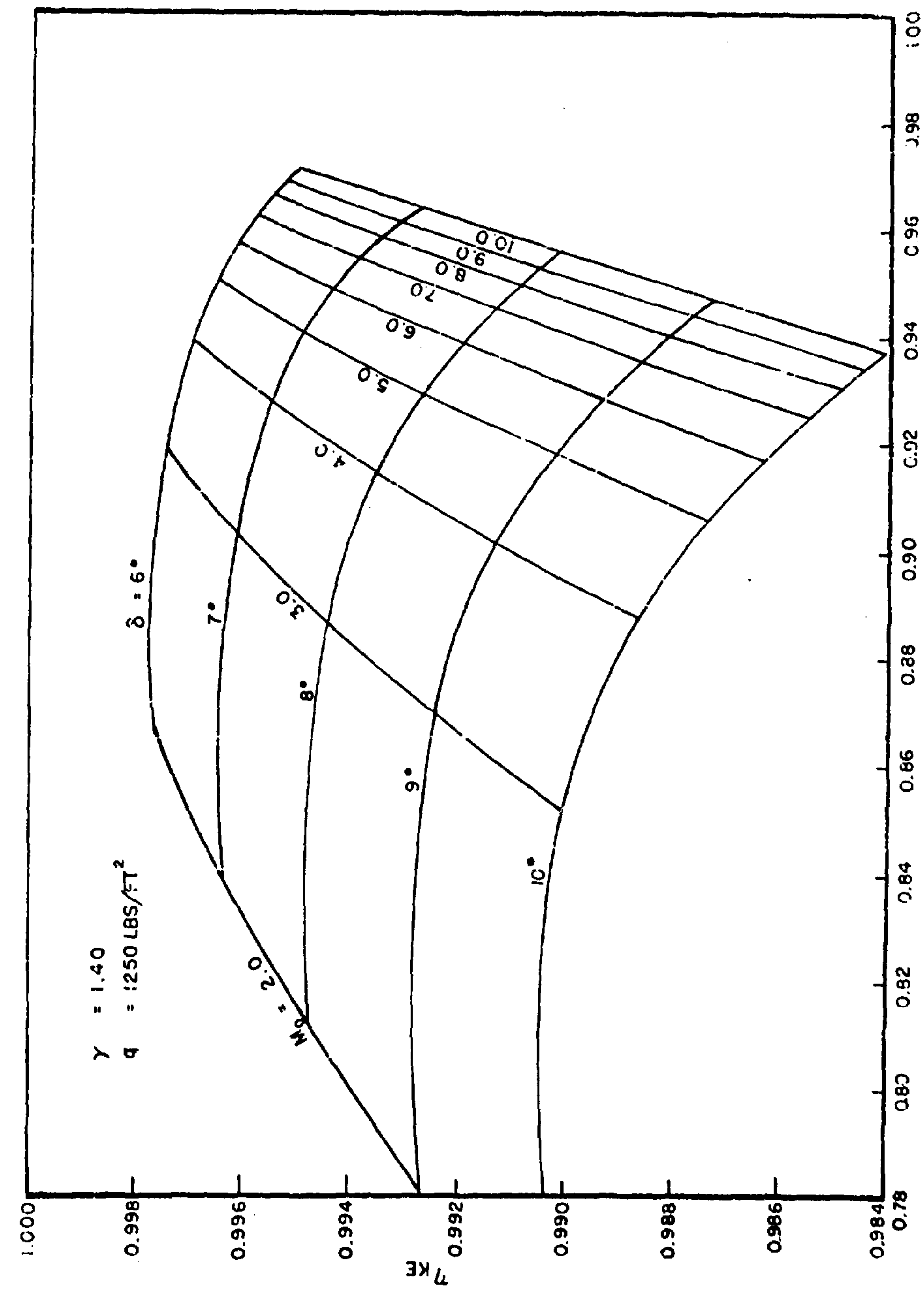

Figure 12. Performance of Two-Shock Intakes at Low Speeds

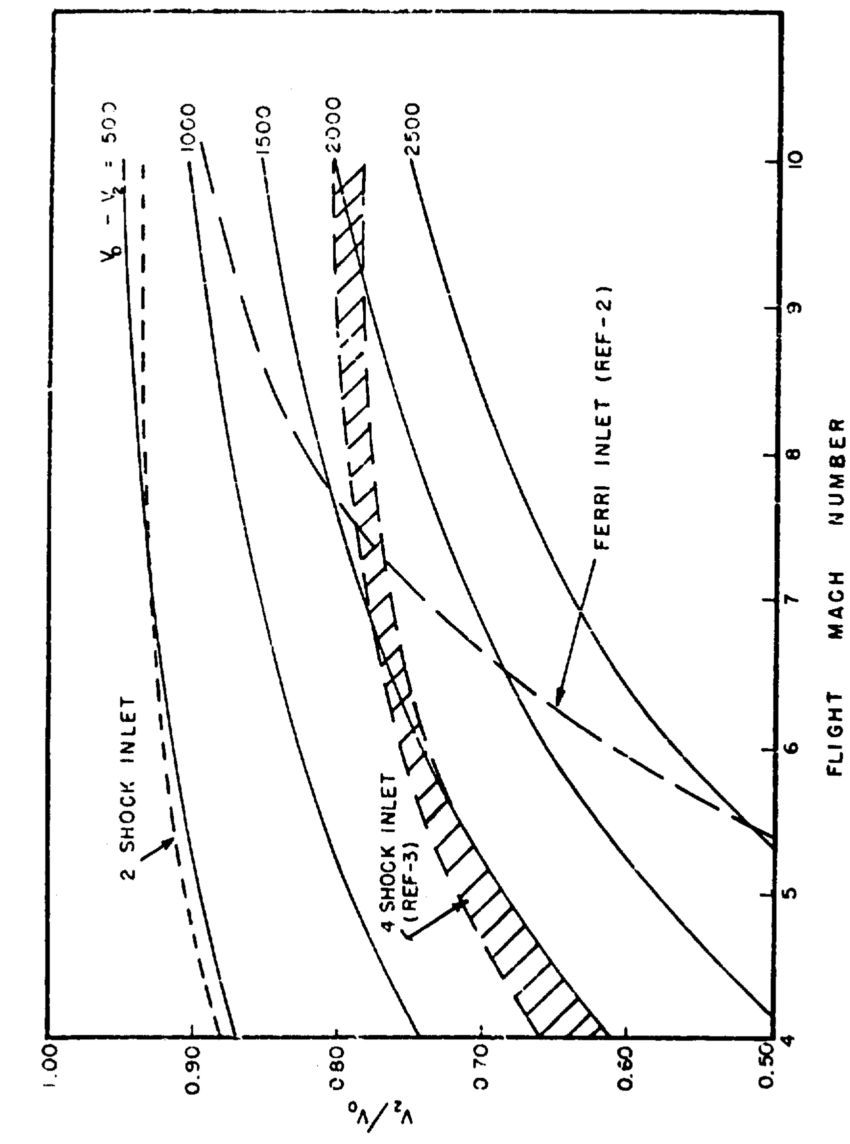

Figure 13. Velocity Ratio for Fixed Velocity Increment Inlets

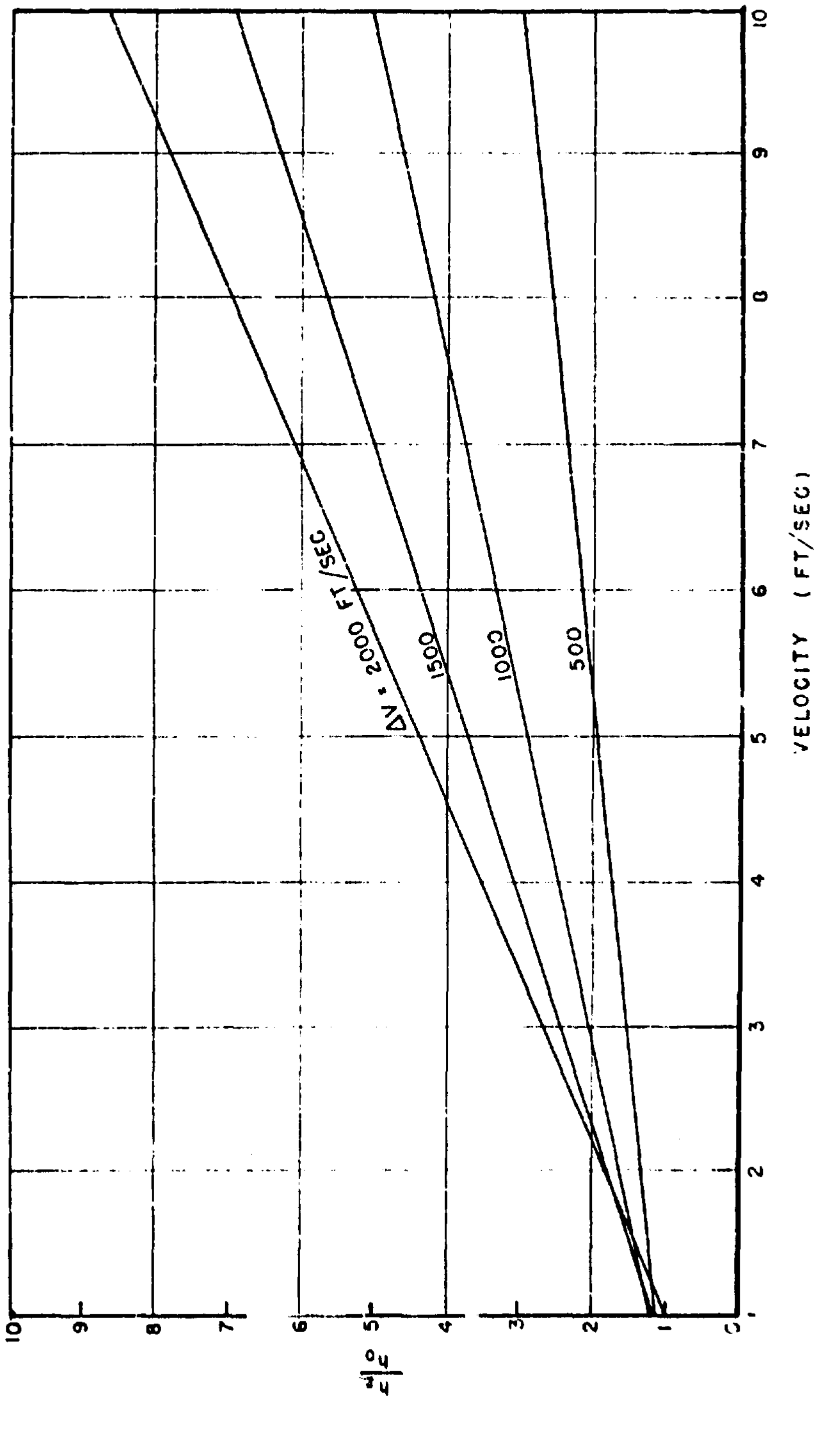

Figure 14. Enthalpy Ratio versus Velocity for Various Values of $V_a - V_s$

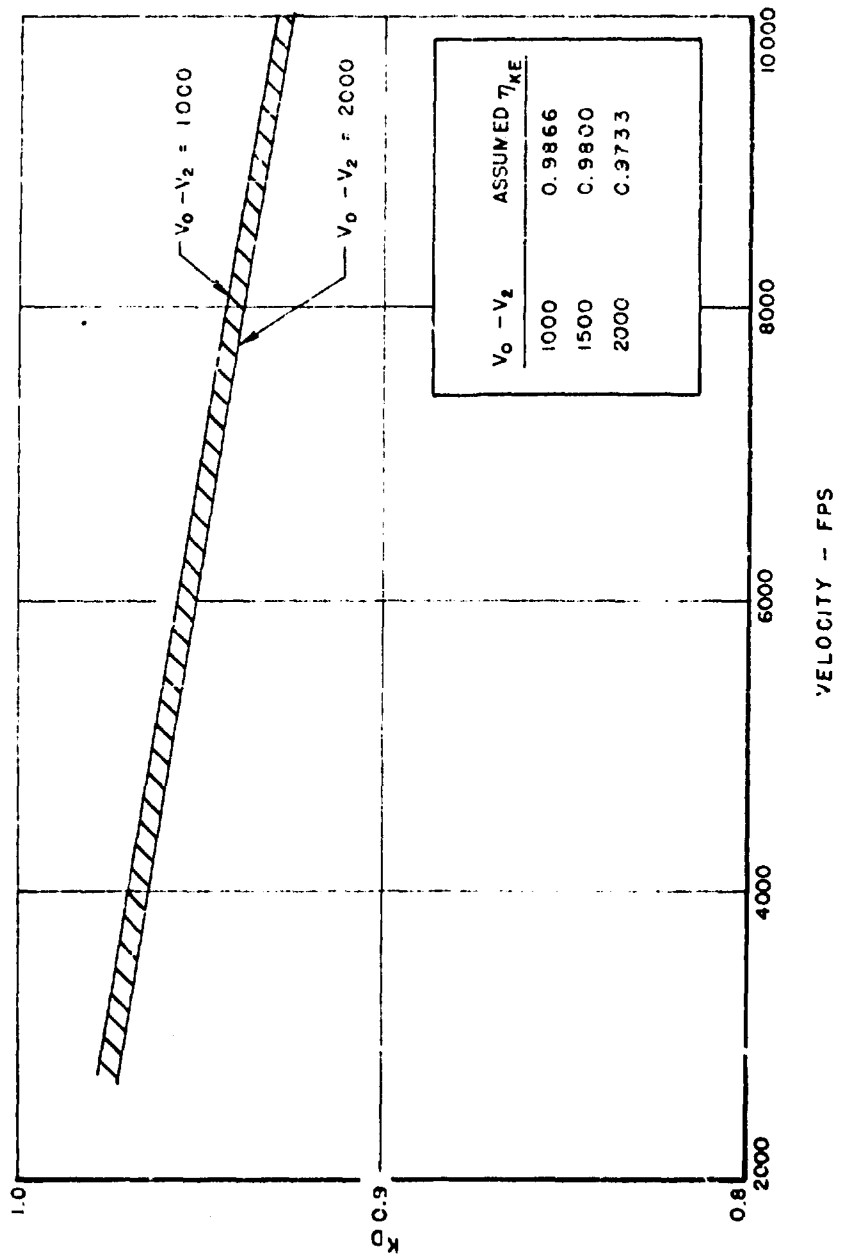

Figure 15. K_D Variation with Velocity for Assumed Variation of η_{KE}

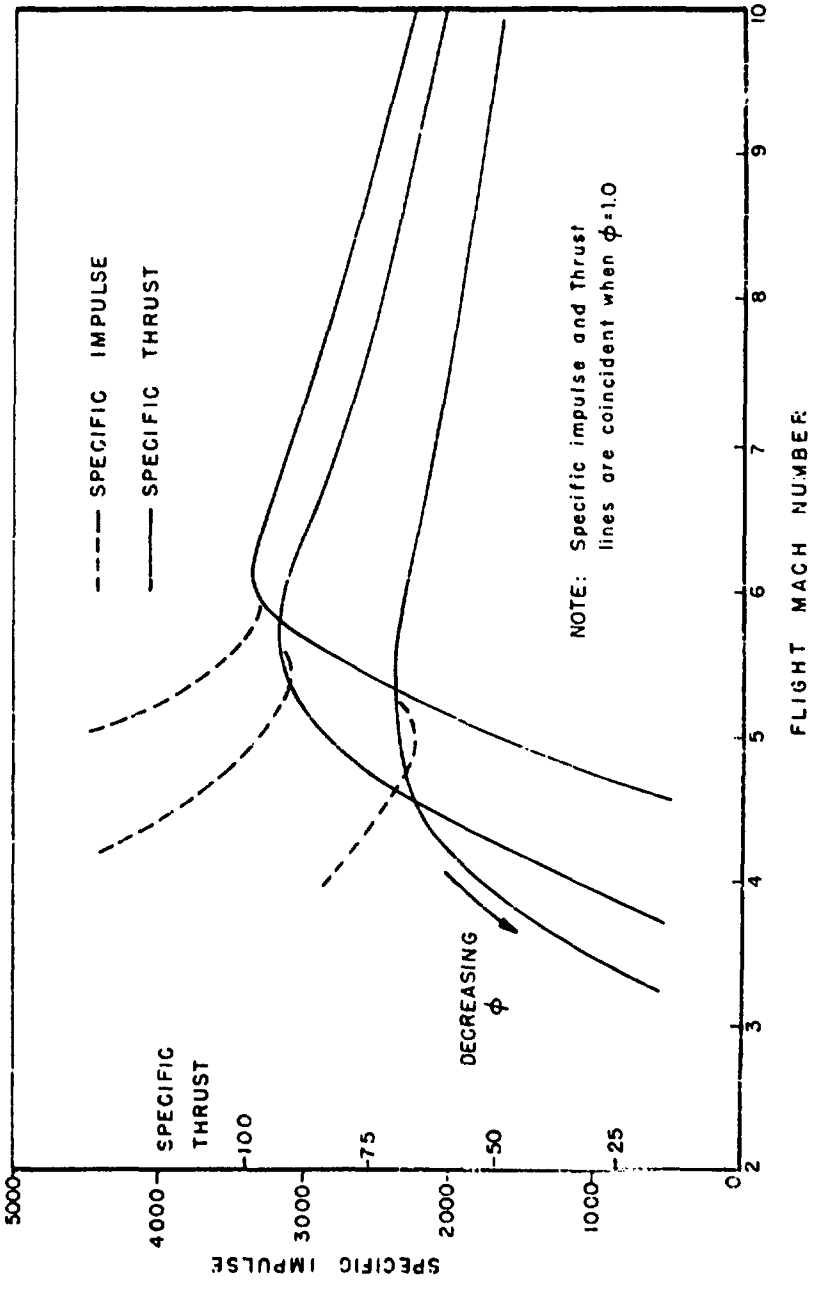

Figure 16. General Engine Performance with Various Diffuser Velocity Increments

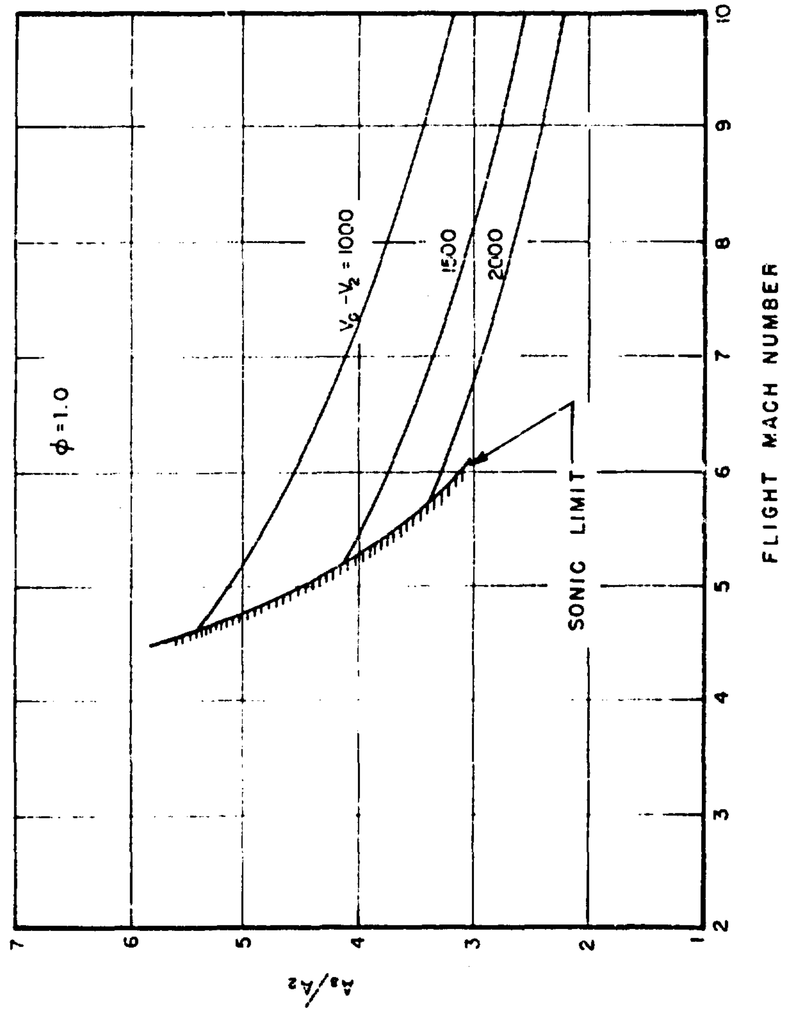

Figure 17. Combustor Area Ratio for Fixed Velocity Increment Inlets

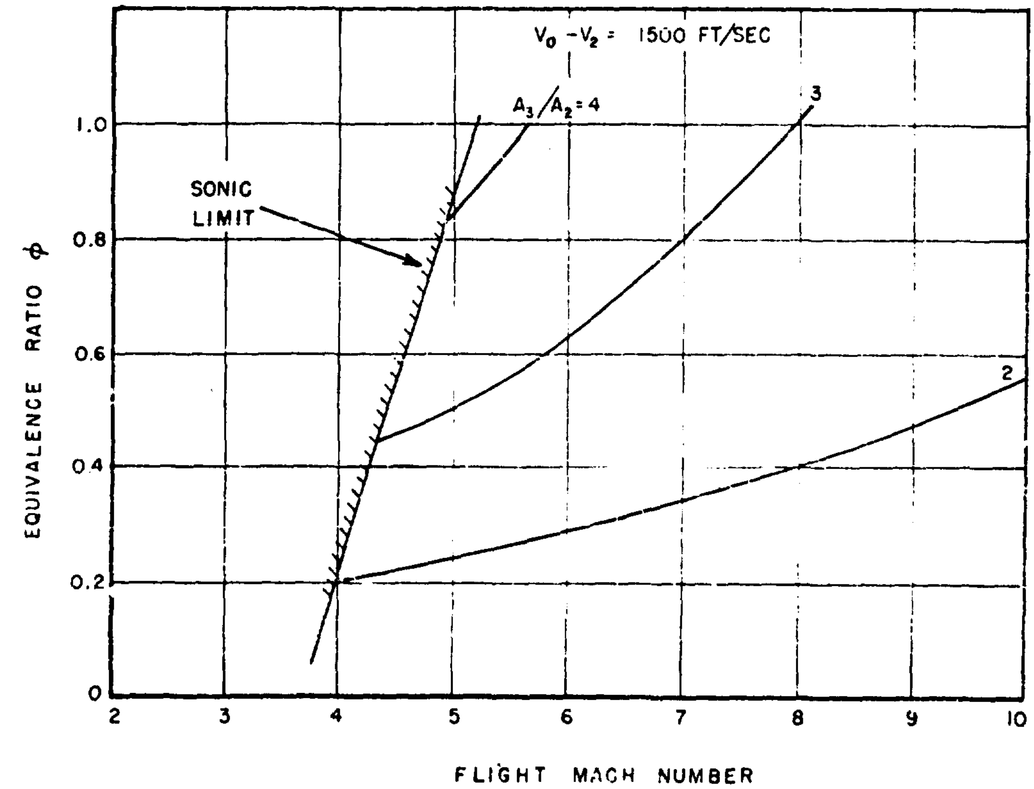

Figure 18. Equivalence Ratio for Fixed Velocity Increment Inlets

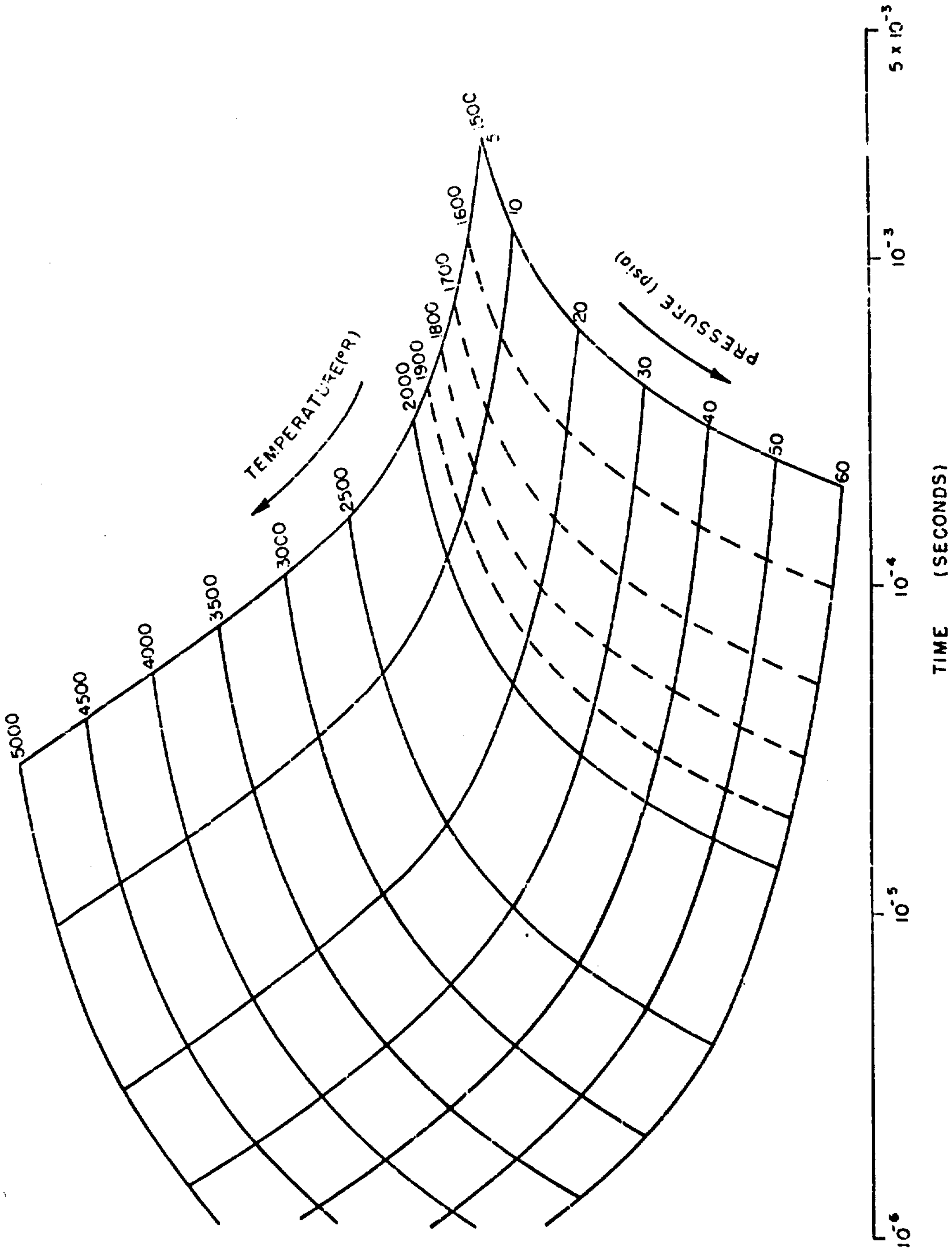

Figure 19. Time Required to Complete Reaction

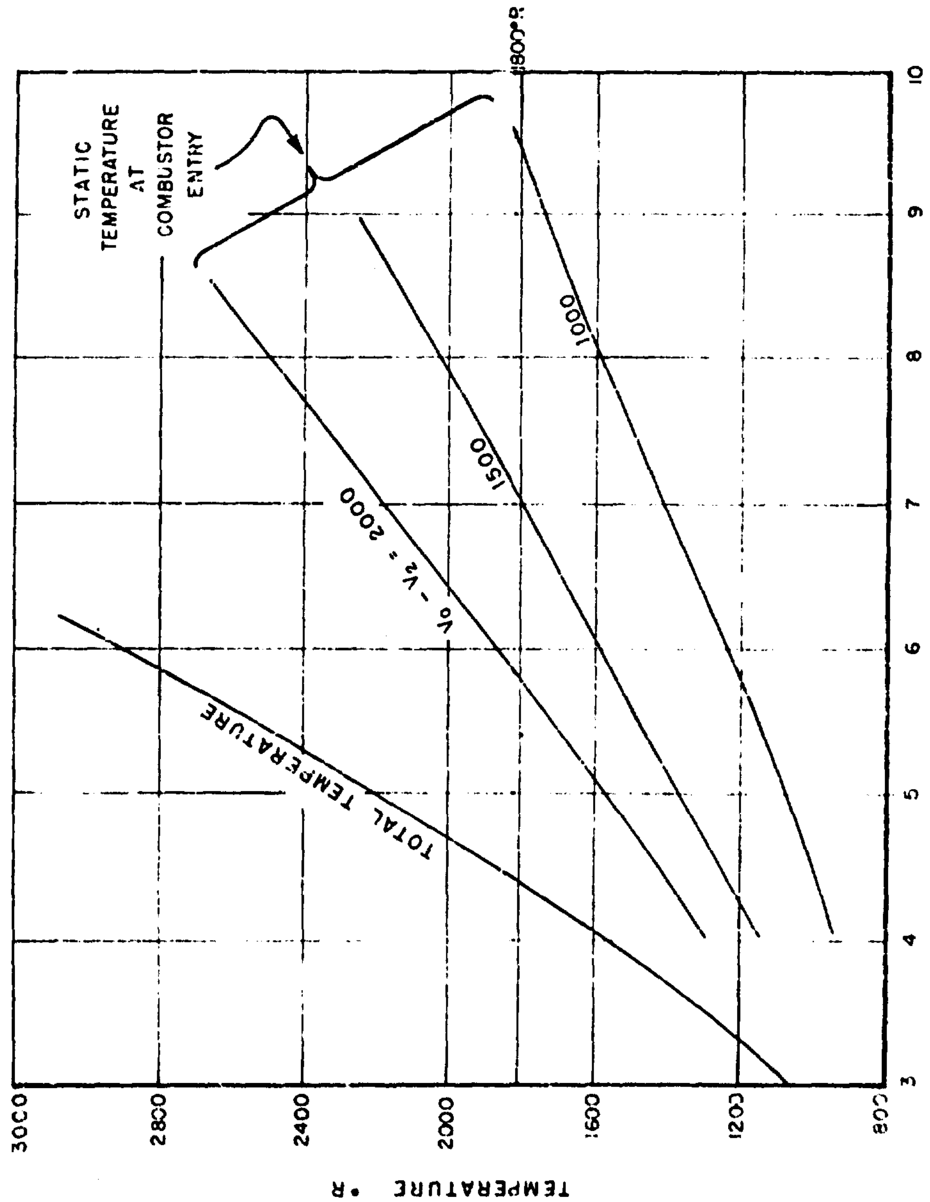

Figure 20. Temperature at Entry to Combustor

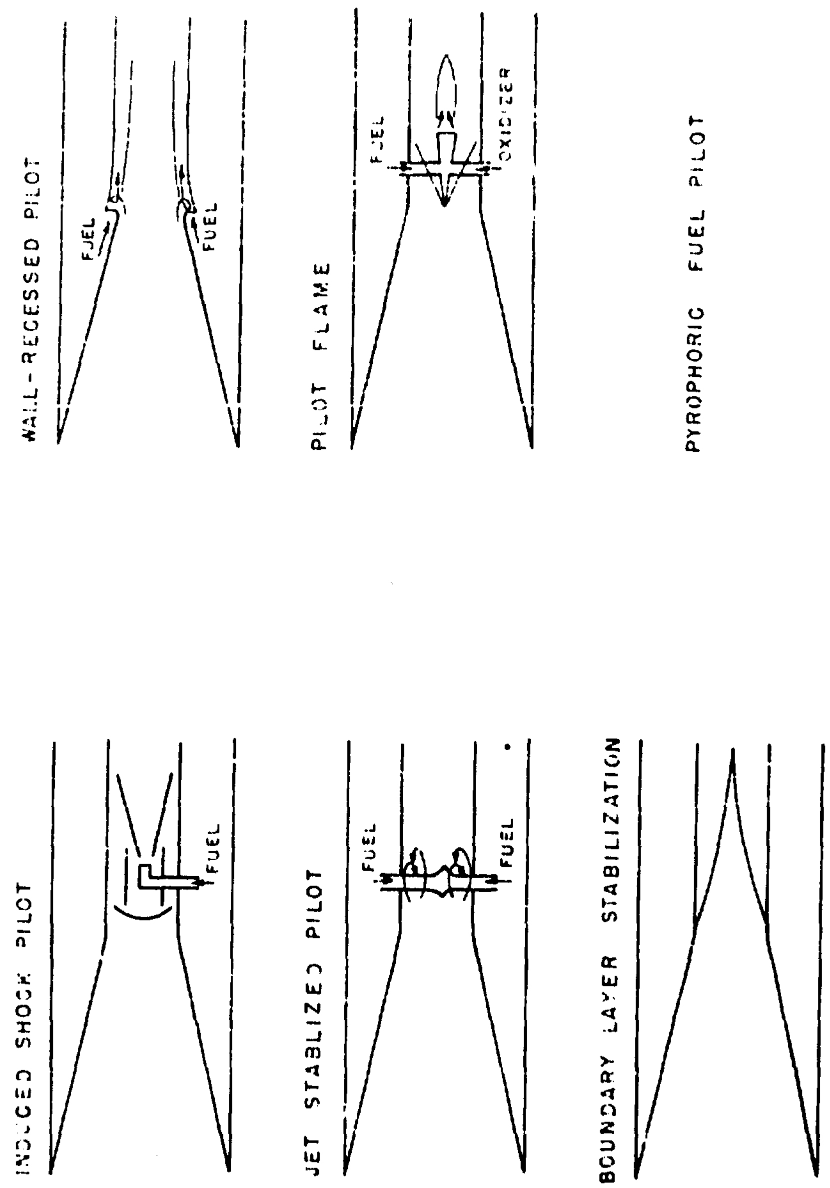

Figure 41 Flame Stabilization

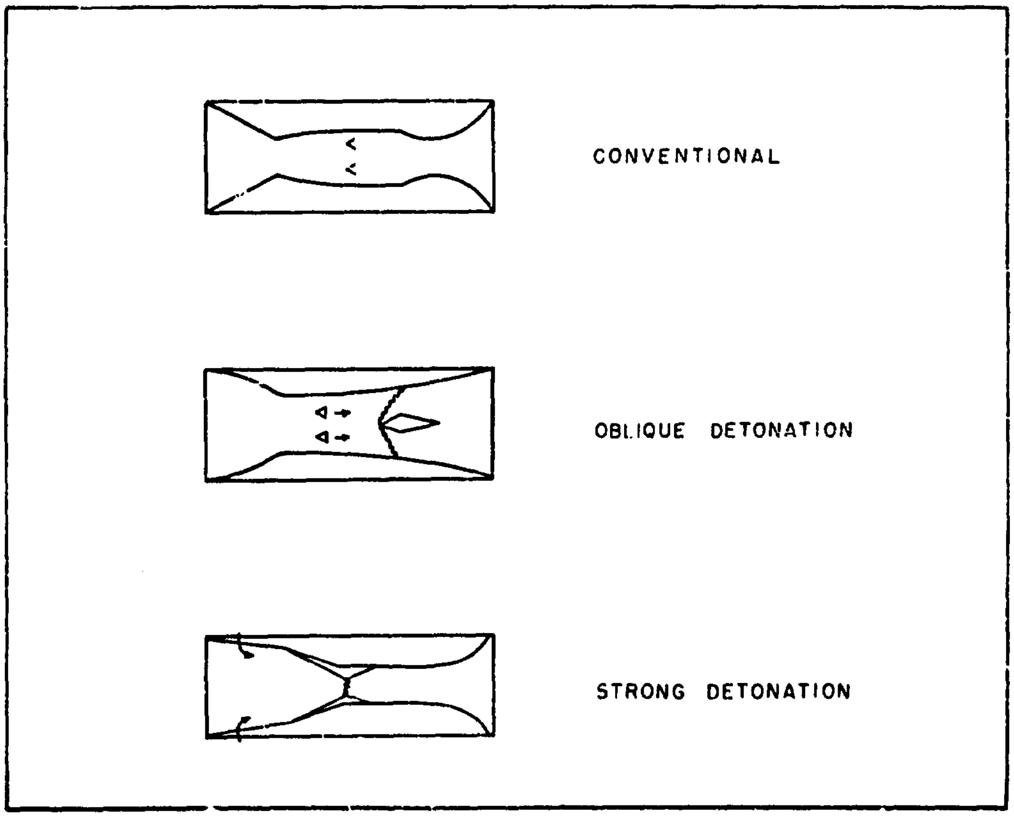

Figure 22. Subsonic Modes of Heat Addition

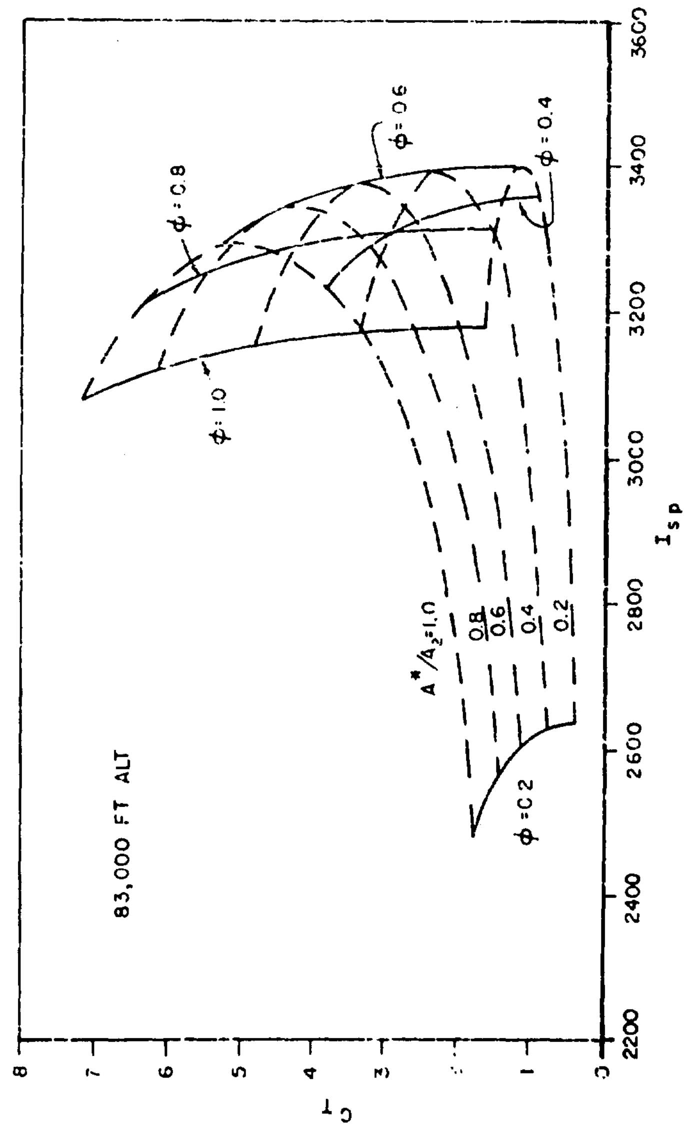

Figure 2[illegible]. Subsonic Combustion Ramjet Performance at Mach 5.0

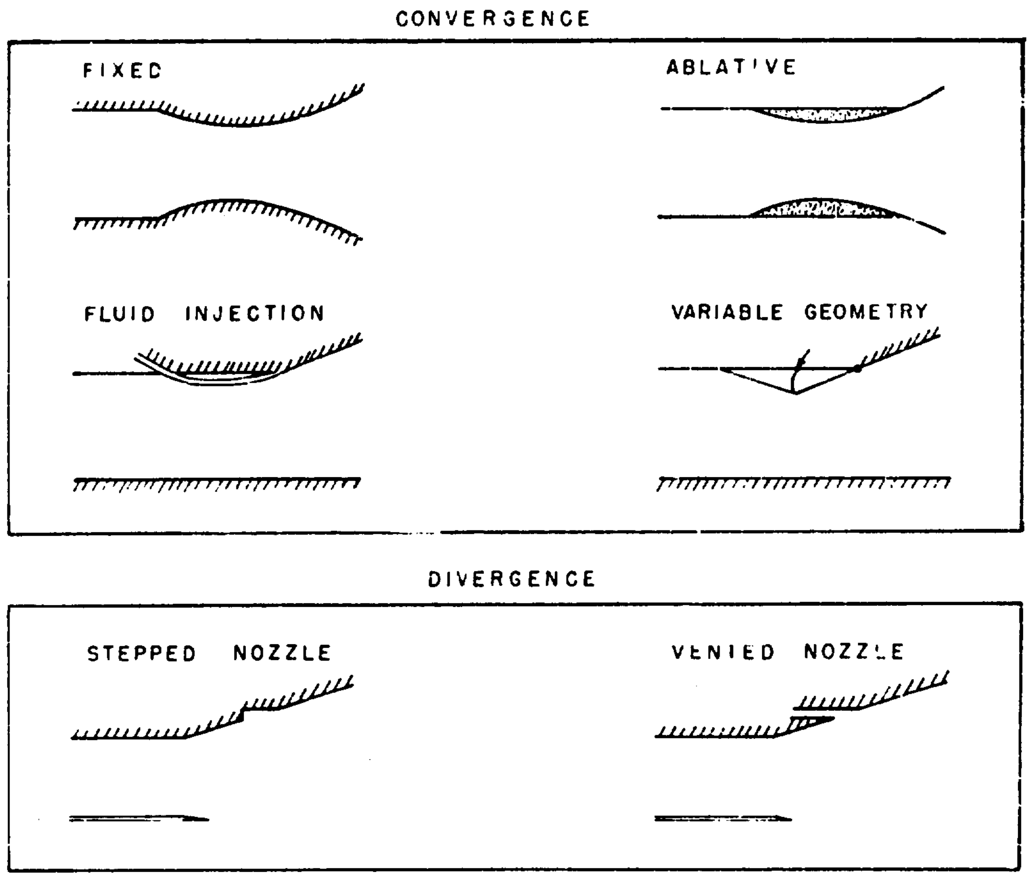

Figure 24. Nozzle Geometry

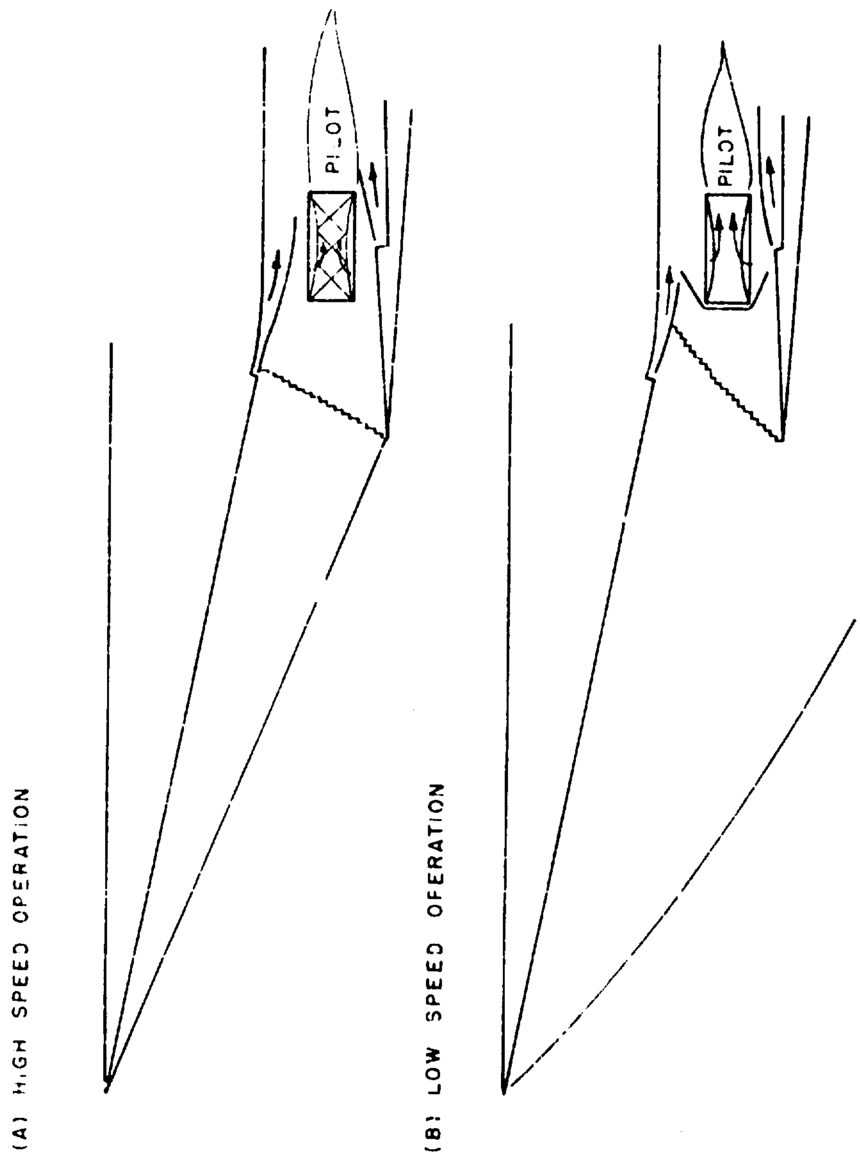

Figure 25. Mixed Flow Engine

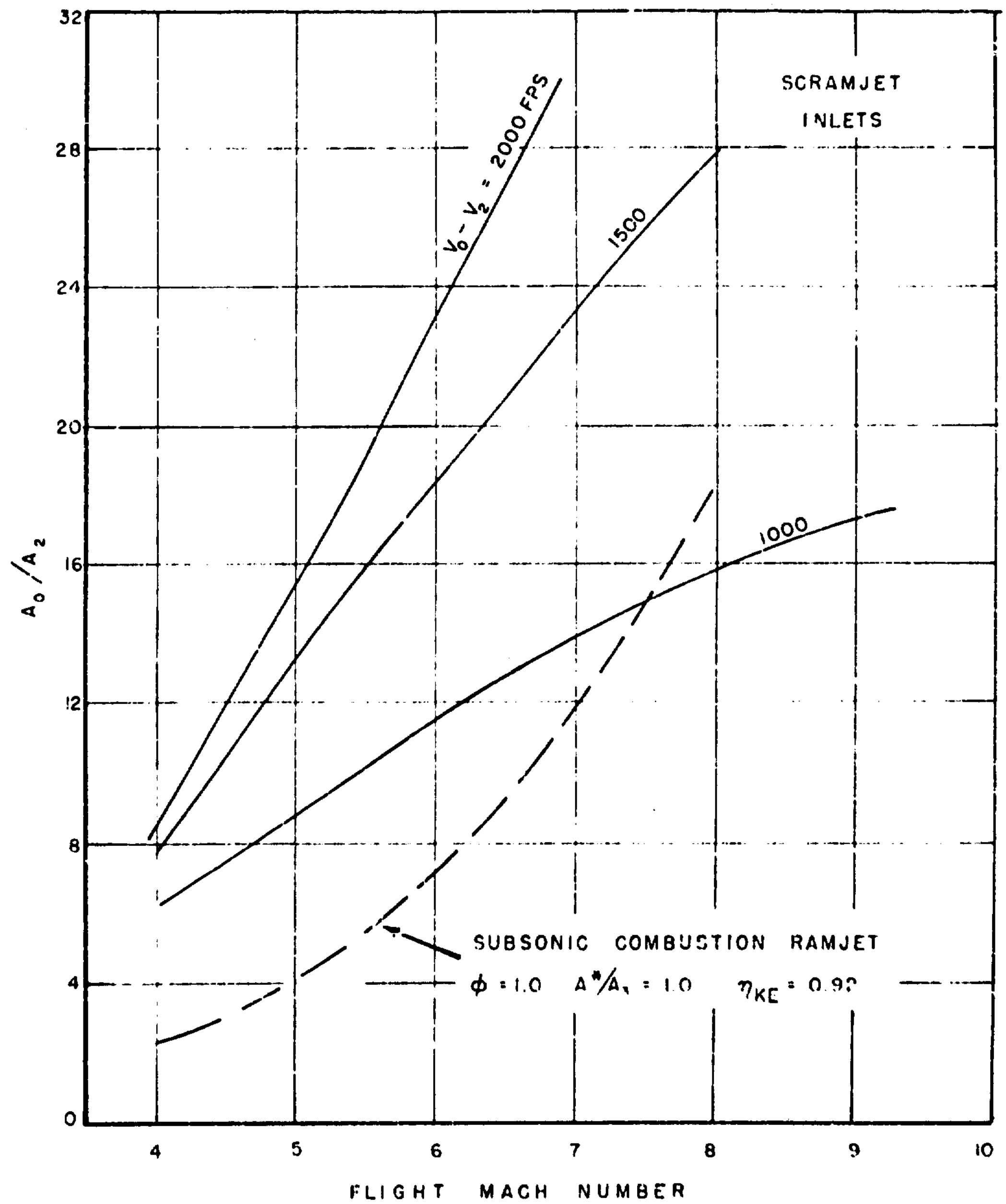

Figure 26. Capture Area Ratios Required for Scramjet and Subsonic Operation

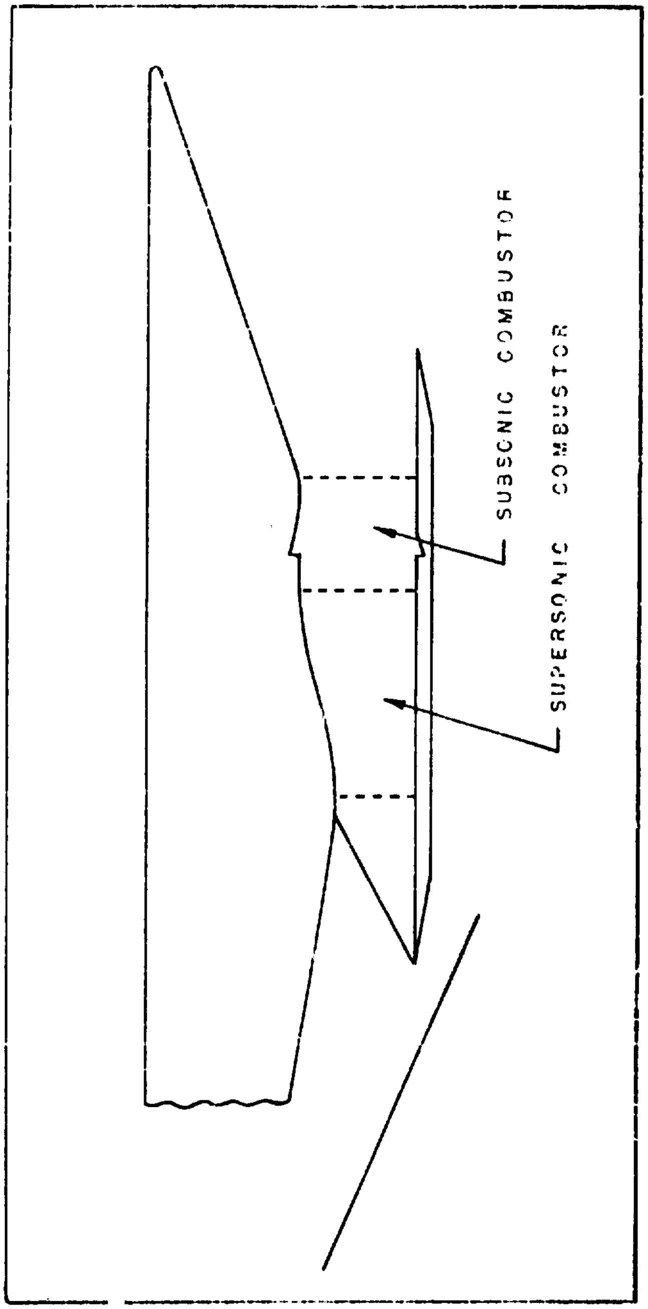

Figure 27. Dual-Mode Combustor - Schematic Arrangement

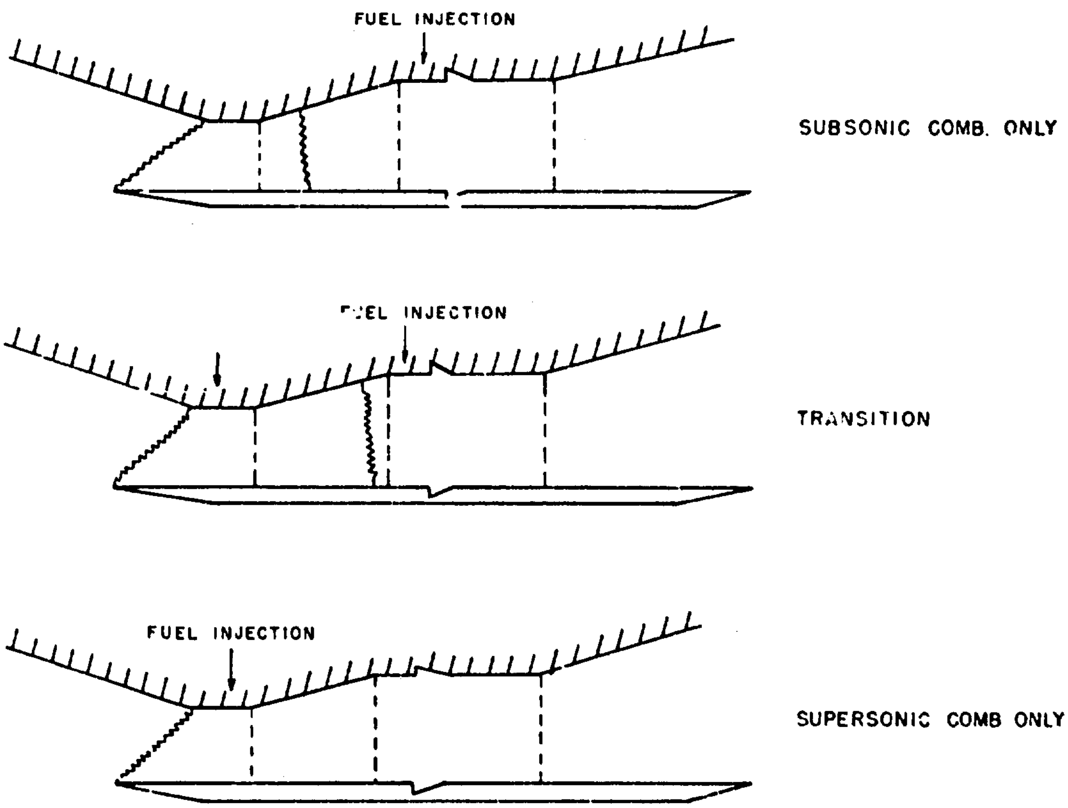

Figure 28. Dual-Mode Combustor - Transition Process

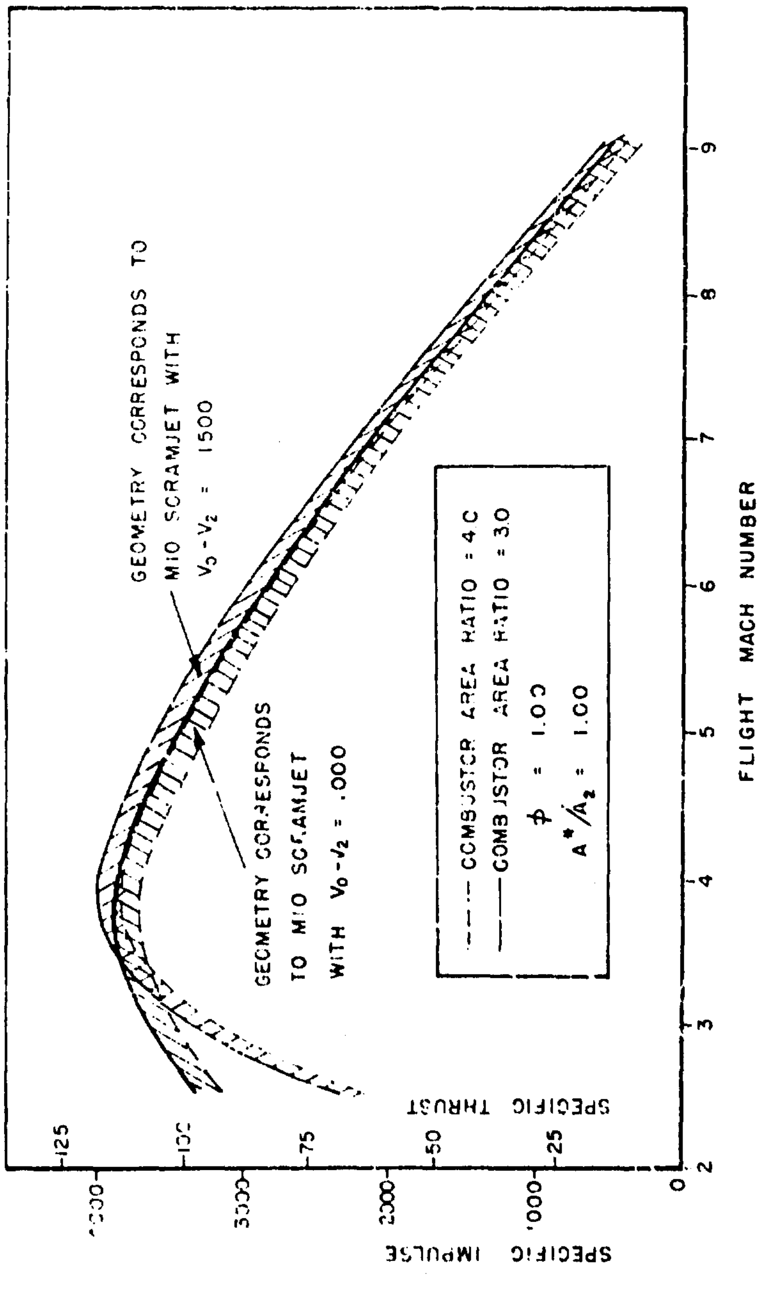

Fig. Performance of Subsonic Burning Mode

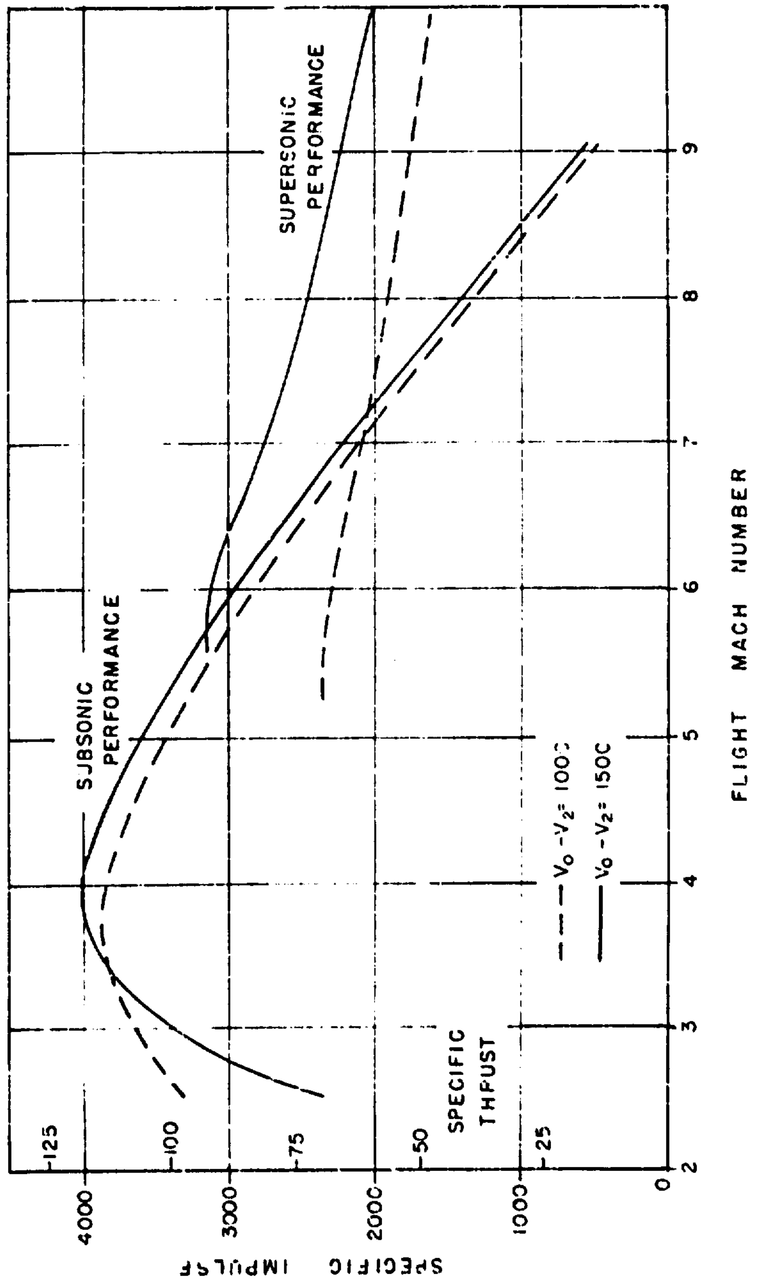

Figure 39. Performance of Subsonic and Supersonic Mode